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**INVESTIGATION OF SCAVENGING SYSTEMS FOR
REMOVAL OF ROCKET EXHAUST GASES
FROM SPACE SIMULATION CHAMBERS**

J. A. Morris and D. E. Anderson

ARO, Inc.

August 1966

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FOREWORD

The research reported herein was sponsored by the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under Program Element 62405184, Project 6950, Task 695001.

The results of the work presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of AEDC under Contract AF40(600)-1200. The research was conducted from November 1964 to October 1965 under ARO Project No. SW2302, and the manuscript was submitted for publication on May 2, 1966.

The authors gratefully acknowledge the assistance of Dr. Louis Brown and Jim Chaille, Rohm and Haas Company, for providing the solid-propellant rockets used in this study; and the assistance of Murl Culp and Bob Prozan, Lockheed Aircraft Corporation, in providing the jet wake computer program, the theoretical basis for this study.

This technical report has been reviewed and is approved.

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ABSTRACT

Tests have been conducted on a rocket exhaust scavenging system as part of a program to develop new techniques for testing at simulated space conditions. Detailed plume characteristics were computed, and a model scavenging system was tested for both cold and hot gas rockets. Results indicate that for the firing of a typical small control rocket it will be possible to maintain 1×10^{-4} torr pressure in a space chamber.

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NOMENCLATURE

\dot{m}_c	Mass flow into vacuum chamber, torr ℓ /sec
\dot{m}_d	Mass flow into scavenging duct, torr ℓ /sec
\dot{m}_r	Gas generator mass flow, torr ℓ /sec
p_1	Pressure in plume upstream of any shocks, torr
p_2	Pressure in plume upstream after a normal shock, torr
p_c	Pressure in vacuum chamber, torr
p_d	Pressure in scavenging duct, torr
p_r	Pressure in gas generator chamber, psia
p_s	Pressure in jet plume, psia
q	Flow into a vacuum chamber, torr ℓ /sec
s	Pumping speed of vacuum system, ℓ /sec
t	Time, sec
v	Vacuum chamber volume, ℓ

SECTION I INTRODUCTION

Various techniques for simulating different aspects of space environment have been developed over the past few years. Some of these techniques have been developed to test specific components or subsystems, and additional techniques and combinations of test methods have been developed for complete systems tests. The work reported herein is concerned with techniques which can be applied to the testing of small rocket engines under space conditions.

Previous work at AEDC in this field was reported in Ref. 1. That portion of the program was devoted to techniques which can be used to remove the rocket exhaust gases by cryogenic and cryosorption pumping.

These pumping techniques can be used to great advantage when plume investigations are to be conducted. However, when it is desired to conduct a systems test on a complete space vehicle, that is, a vacuum-thermal test during which time the control rockets must be fired, adequate simulation cannot be maintained by allowing the gas to fill the entire space chamber. This is illustrated in Fig. 1a for a 100-lb rocket in a large space chamber. When the rocket is fired, the pressure in the chamber rises to such a level that thermal balance on the test vehicle is changed from what it would be in space. Although there is some indeterminacy because of different types of vehicles as to the exact pressure which must be maintained to keep gas conduction effects low, for this discussion 1×10^{-4} torr will be used as the upper limit of acceptable pressure. It can be seen that a period of over 10 min would be required before the pressure returned to an acceptable level.

This difficulty can be prevented by removing a major portion of the rocket exhaust gas through a plume scavenging duct as illustrated in Fig. 1b. For equivalent engines and after removing 95 percent of the engine exhaust through the duct, the period of incorrect thermal balance can be greatly reduced or eliminated. The external pump may be a conventional mechanical pump because the exhaust scavenging system inlet can be designed to recover some of the kinetic energy of the rocket exhaust gases.

This report presents analytical and experimental results which were obtained in order to determine the effectiveness which could be expected by using a plume scavenging system to maintain the space chamber pressure at a low level during engine firing.

SECTION II SYSTEM CONSIDERATIONS

2.1 STATEMENT OF PROBLEM

The general problem can be stated as requiring that the exhaust from a control rocket should be captured by an inlet placed near the nozzle. This inlet should capture the desired amount of gas and should give as much pressure recovery of the exhaust as possible in order to facilitate its removal from the chamber. There are several elements and constraints of this problem which must be considered. The elements of the problem are:

1. Calculation of the rocket exhaust plume and the decision as to what percentage of the plume can be removed by the chamber pumping system and the amount which must be removed by the inlet.
2. Choice of the type of inlet which is used and its ability to capture the desired exhaust products.
3. The pressure recovery which can be expected from the inlet.
4. The effect of rocket shutdown which may allow gas to spill back into the space chamber from the scavenging duct.

The major constraints which were imposed include:

1. The inlet should be far enough from the nozzle so that the jet can expand freely for some distance; however, the inlet must be reasonably small so that large radiation blockages do not occur. It is desirable to allow the jet to expand so that heating from the jet gases can be experienced by the space vehicle surfaces. In most cases this means the inlet should be about 20 nozzle exit radii downstream of the exit. In addition, since the boundaries of the plume are rather poorly defined at these pressures (1×10^{-4} torr) and contain lower energy gas, no attempt would be made to capture over 94 percent of the gas into the inlet.
2. The inlets should be of relatively simple fixed geometry. If liquid-nitrogen (LN₂) cooling is desirable, it could be used on the inlets.
3. Hot and cold jets would be used in the test program. The exhaust products should be similar to those expected in test programs; thus, they would contain H₂, N₂, CO, CO₂, and H₂O as major constituents.

4. The experimental program would be carried out in a 7-ft-diam vacuum chamber which then fixed the maximum size of the rockets to be tested in the experimental portion of the program.

2.2 APPROACH

Since the chamber in which the experimental work was conducted is the 7-ft-diam Aerospace Research Chamber (7V), the maximum size of gas generator or rocket which could be used was established. The largest size practical was desired so that scale ratios from the experiment to application on larger tests (~100-lb thrust engines) would not be too large. Thus, the rocket size chosen was approximately 5-lb thrust and a corresponding cold gas generator of the same size. A test setup, shown in Fig. 2, was then chosen. With this type of test setup, it was possible to pump the major portion of the gas flow out through the scavenging duct with an external pump and to pump the bypass or spilled flow with the test chamber pumping system.

Computations were then performed which defined the plume properties which could be expected, and it was decided that the inlets would be designed to capture 94 percent of the nozzle flow. Three inlets were then designed incorporating different degrees of complexity and different anticipated levels of recovery efficiency. The recovery of the inlets was estimated, and instrumentation was placed in the gas generator and rockets, the vacuum test chamber, and scavenging systems so that the system performance could be measured. Tests were then run with a cold gas generator, and the percent captured by the inlets and the pressure recovery of the inlets were measured. Since a 5-lb liquid rocket was not available, a solid rocket with a 3/4-sec burn time was used. The amount captured by each inlet was measured using the solid rockets. Since the burn time was short, maximum recovery with the hot gas could not be obtained. The effect on the chamber pressure at rocket shutdown was determined by operating with and without a reverse flow control in the inlets.

2.3 RELATED WORK

A considerable amount of analytical and experimental work has been conducted in plume expansions, ejector-diffusers for engine tests, supersonic inlets, and cryopumping or cryosorption of various gases.

A summary of plume studies under low pressure conditions is presented in Ref. 2. This work demonstrated the applicability of analytical techniques to predict plume properties. A more detailed, characteristics

solution, computer program was developed and reported in Ref. 3. This program has been adapted for use at AEDC, and Ref. 4 reported good experimental agreement with properties computed by this program. It was this computer program which was used to predict the plume properties for the cold gas jets and rockets used in these studies.

Work on ejector-diffusers which are used to improve the altitude performance of rocket engine test facilities has been carried on for some time at various industry and government test centers. Some of this work is described in Ref. 5. A study of ejectors without secondary flow is reported in Ref. 6. Work in which a small part of the plume was spilled is reported in Ref. 7. This present program was intended to have the diffuser spill a significant portion of the plume boundary into the chamber to be pumped by the chamber pumping system, thus extending the data obtained to date. In addition, it was desired that the chamber pressure be about 1×10^{-4} torr, whereas previous results have been with chamber pressures above 10^{-2} torr. However, some of the analytical techniques which have been developed (Ref. 6) could be used to estimate system performance.

A large background of work has been done on supersonic inlets (Ref. 8) and supersonic wind tunnel diffusers (Ref. 9). This work has been related to essentially uniform inlet conditions and is difficult to apply without the development of a large computer program because of large nonuniformities in the present case. However, certain generalities may be ascertained, in particular the fact that the pressure recovery in most simple supersonic diffusers can be estimated by using the normal shock pressure recovery based on diffuser inlet Mach number, and that for straight duct-type diffusers several duct diameters are required in which the pressure recovery can occur.

In recent years, work on cryopumping and cryosorption has proceeded at an increasing pace. As noted in Section I, early work on this program was devoted to techniques of cryopumping and cryosorbing rocket exhaust products (Ref. 1). Several additional studies have been carried out on pumping of rocket exhausts and individual gases (Refs. 10 and 11). When the engines to be tested are larger than 100-lb thrust, the refrigeration required to condense CO_2 , N_2 , and H_2 becomes prohibitive. For the program under study here, cryopumping would only be used for spillage flows. The information in Refs. 1, 10, and 11 can be used to estimate chamber pumping speed for the spillage flows and hence chamber pressure which will be experienced during rocket firing.

2.4 SYSTEM ANALYSIS

2.4.1 Plume Geometries

As stated, the jet plumes were computed using a method of characteristics network. The information required to construct the network consists of the nozzle geometry, the gas properties, and the pressure ratios which can be expected between the nozzle total pressure and the test cell pressure. Typical results for a cold jet and a rocket of the type used in this study are shown in Figs. 3 and 4. As would be expected, some difference exists between the two, and if an inlet is expected to capture the same percentage of cold and hot gas it must be located to intercept equivalent mass flow lines in each case. Since one of the constraints was that the gas would be allowed to expand for about 20 nozzle radii, it can be seen that the inlet Mach numbers will vary from 9 to 20 for the cold jet and from 7 to 15 for the rockets. Variations in plume boundary with chamber pressure are illustrated in Fig. 5. Thus, for the conditions used in this test (i. e., rocket chamber total pressure ~1000 psi) it can be seen that the chamber pressure must reach $\sim 10^{-1}$ torr before the 95-percent mass flow line is influenced. The static pressure near the point where the outer lip of the inlet would be located is $\sim 10^{-2}$ torr; thus, the entire flow field between the nozzle and the lip is in the continuum flow regime.

2.4.2 Inlets

Three inlets designed for use in the program were: (1) a straight cylindrical duct, (2) a duct containing a conical centerbody, and (3) an inlet which attempted to gain the additional pressure recovery by approximating an isentropic compression inlet. These inlets were designed to capture 94 percent of the nozzle flow. This made their lip location occur at 21 nozzle radii for the cold gas case and at 15 nozzle radii for the hot gas tests.

The pressure recovery for the inlets could be estimated using techniques presented in Refs. 9 and 10; however, because of the large flow nonuniformities, the flow field must be divided into a large number of segments and an extremely large computer program results. In view of this large effort, these computations were not attempted at this time. By making some mass weighted averages of the normal shock pressure recovery, an estimate was obtained for the pressure recovery of the system. Thus

$$\sum_{n = \text{centerline}}^{n = \text{lip}} \frac{\dot{m}_n}{\dot{m}_d} \left(\frac{p_2}{p_1} \right)_n \left(\frac{p_1}{p_r} \right) = \frac{p_d}{p_r} \quad (1)$$

where

\dot{m}_n	is the mass flow in the n^{th} segment
\dot{m}_d	is the mass captured by the inlet
$\left(\frac{P_2}{P_1}\right)_n$	is the normal shock static pressure recovery associated with the n^{th} segment
$\frac{P_1}{P_r}$	is obtained from the expansion

gives an estimate of the pressure which can be expected in the scavenging duct. The corresponding pressure ratio obtained by this technique for the cold gas and rockets is 2.4×10^{-4} and 2.8×10^{-4} , respectively. Data from Ref. 7 indicate that the straight duct should be expected to operate at about this pressure recovery. Data from hypersonic inlets indicate single angle compression cones should provide 1.5 to 2 times the pressure recovery of a straight duct (Ref. 10); thus, this configuration is expected to have p_d/p_r values of 3.6×10^{-4} and 4.2×10^{-4} for cold gas and rockets, respectively. The third configuration might be expected to have a somewhat higher compression ratio. However, the problems of off-design operating conditions for the inlets with centerbodies can be expected to be worse than the straight duct.

Since the inlets must be matched to an existing scavenging pumping system or their operating characteristics used to determine an appropriate pumping system, information, illustrated in Fig. 6, must be obtained from the test. Since the inlet is immersed in a hypersonic stream, it can be expected to function as a normal inlet where back pressures (p_d) up to a certain value will have no influence on the amount of gas captured by the inlet. Further increases in the back pressure cause the inlet flow to break down, and gas is spilled around the inlet. The intersection of the inlet characteristic and the pump characteristic will determine the operating point for a given system. For a pump which is too small, the inlet will spill too much gas in the test chamber causing incorrect thermal-vacuum results. For an oversized pump, test results will be satisfactory but the most economic system is not obtained.

2.4.3 Test Chamber

The pumping speed of the test chamber had been measured previously, and the results are presented in Fig. 7. These data were taken for individual room temperature gases admitted into the chamber through a diffuse emitter. With the assumed spillage flows it is possible to compute

the pressure-time response expected in the chamber by using the standard pumping speed equation (Ref. 13).

$$p_c = \frac{q}{s} \left(1 - e^{-\frac{s}{v}t} \right) \quad (2)$$

However, the previous data did not apply very well to hot gas tests, and a new average pumping speed was required. This could be obtained by firing the rockets directly into the chamber with no scavenging and computing s from the above equation and the pressure-time trace. Then this pumping speed could be used in the computation of q from measured values of p_c when the scavenging system is being used. Thus, the value of $\frac{\dot{m}_c}{\dot{m}_r}$ for the same value of t is proportional to p_c/p_c' , where p_c' is the pressure obtained with no scavenging system. Since

$$\dot{m}_r = \dot{m}_c + \dot{m}_d \quad (3)$$

\dot{m}_d/\dot{m}_r can be computed. This procedure could also be used for the cold gas case to check the previous pumping speed measurements, and since a flowmeter was to be used in the scavenging duct and could be read for the cold gas runs, a cross check on \dot{m}_d/\dot{m}_r was obtained.

SECTION III APPARATUS

3.1 ARC (7V)

The ARC (7V) (Fig. 8) is a 7-ft-diam by 12-ft long stainless steel cylindrical space chamber with a stainless steel cryoliner. In these tests, the cylindrical portion of the liner was normally operated at 77°K and the end panels at 20°K. The end panels are cooled with a 1-kw capacity gaseous helium (GHe) refrigerator. Two 32-in. diffusion pumps with LN₂ baffles are attached to the bottom of the chamber.

3.2 COLD AND HOT GAS GENERATORS (THRUSTERS)

The hot gas jet was created by a small solid-propellant charge. The charges were installed in a case as shown in Fig. 9a, and the propellant burn time was about 0.75 sec with a typical chamber pressure curve as shown in Fig. 9b. The gas products were H₂O, CO₂, N₂, CO, and H₂. There were four units mounted in the chamber for each pump-down, and an indexing mechanism was used to align them with the inlets.

The cold gas jet was obtained by connecting a supply tube into the same type of case that was used for the hot gas. The gas used was a

85:15 mixture of N_2 to CO_2 which simulated the percent 77°K condensables for the hot jet. Both the cold and hot jets could be aligned so that they could exhaust directly into the chamber or into the scavenging system.

3.3 EXHAUST SCAVENGING SYSTEM

The exhaust scavenging system consisted of four major parts: the inlets, the CO_2 reverse flow control, the connecting ducts, and the pumping system.

3.3.1 Inlets

The three inlets which were used are shown in Figs. 10a and b, 11a and b, and 12. The inlets could be connected to the connecting duct with or without the CO_2 control in place. Runs with Configuration A did not use the CO_2 control.

3.3.2 CO_2 Reverse Flow Control

The CO_2 ejector was a simple annular ejector with an annular driving nozzle (Fig. 13). The CO_2 driving nozzle operated at 800 psi, and the CO_2 was condensed in the LN_2 -cooled connecting duct. The CO_2 flow in the ejector was controlled by varying the throat area of the driving nozzle.

3.3.3 Connect Duct and Scavenging Pumps

The connecting duct was 6 in. in diameter and contained the flow measuring equipment. The scavenging pump was a 500-cfm mechanical pump. A 16-in. diffusion pump was installed in the line so that backflow into the chamber from the scavenging duct did not occur during pumpdown. During firing the inlet pressure to the pump went above its operating range, and it only served as part of the connecting duct during the runs. In later tests it was found unnecessary and was removed. The connecting duct was about 100 ft in length. In some cases the duct was connected to a 1200-ft³ ballast chamber so that the duct pressure could be held below that which the 500-cfm pump could maintain.

3.4 INSTRUMENTATION

The general arrangement and location of the instrumentation is shown in Fig. 14. Several types of pressure gages were used to cover the rather wide range of pressures encountered. The pressure and

temperature of the gas generators were measured. Pressure, temperature, and flow through the orifice were measured in the scavenging duct. Pressures were measured in the chamber using quadrupole and time-of-flight mass spectrometers and various pressure gages. The output of the gages and thermocouples was recorded on an oscillograph, and the output of the mass spectrometers was recorded with a motion picture camera viewing the output scope.

SECTION IV PROCEDURE

All pressure gages were calibrated in the instrument laboratory prior to test installation. The test sequence consisted of: installing the desired inlet configuration, loading the hot gas generators and connecting the cold gas generator, evacuation of the chamber to a pressure below 1×10^{-6} torr, in-place calibration of the pressure gages, re-evacuation of the chamber to below 1×10^{-6} torr, and cooling of the desired cryosurfaces. Then the gas generators were operated and measurement of the operating parameters was made; a post-test in-place calibration of the gages was made if any discrepancies were noted. After this procedure, the chamber was returned to atmosphere.

The data obtained during the tests was plotted versus time. With these data, it was then possible to compute the mass of gas captured by each inlet (\dot{m}_d) using Eqs. (2) and (3) as well as the measured flow through the orifice. The duct pressures (p_d) corresponding to these mass flows were measured, and with this information it was possible to plot duct mass flow versus duct pressure for the different inlets. The data were normalized by dividing the duct mass flow by the total mass flow (\dot{m}_d/\dot{m}_r) and the duct pressure by the rocket chamber pressure giving (p_d/p_r).

SECTION V RESULTS

5.1 COLD GAS TESTS

5.1.1 Typical Cold Gas Runs

A typical cold gas run with Configuration A is shown in Fig. 15. The gas supply pressure, chamber pressure, and duct pressure illustrate the response of the system and the steady-state portion of the run

with a fixed setting of the throttle valve. Normally attempts to run the gas for periods over 10 sec would cause the 20°K end panels to warmup so that their cryopumping efficiency would fall off and the chamber pressure would rise.

Data were taken with the throttle valve at different positions from fully open to closed, thus varying the scavenging duct pressure. When the throttle was closed during a run, the results shown in Fig. 16 were obtained. Here it can be seen that as the valve closed, the scavenging duct pressure would start to increase. During the early part of the increase, no effect is noted on the chamber pressure indicating no change in the amount of gas captured by the inlet. Further increases in duct pressure caused the inlet to spill additional amounts into the chamber as evidenced by the rise in chamber pressure.

5.1.2 Summary of Cold Gas Results

A summary of the cold gas results is presented in Figs. 17a, b, and c for inlet Configurations A, B, and C, respectively. These plots were obtained as described in Section IV. For the different configurations, the maximum pressure recoveries (p_d/p_r) were about 2.5, 2.9, and 3.5×10^{-4} . Since the rocket pressure (p_r) was 750 psia or 3.75×10^4 torr, this corresponds to duct pressures of 9.4, 10.9, and 13.1 torr for the three inlets.

5.1.3 Effects of Cooling on Cold Gas Results

The cooling of the inlets to 77°K with LN₂ had little effect on the maximum pressure recovery which could be obtained by the inlets. The cold inlets and duct did condense a large portion of the CO₂ in the gas, thus reducing the external pumping required. This effect can be seen in Fig. 18 for a fixed throttle valve setting. Cooling the system had almost no effect on the chamber pressure; however, the duct mass flow and hence duct pressure were down because of the removal of the CO₂.

5.2 HOT GAS TESTS

5.2.1 Typical Rocket Firing

The results of two typical rocket firings are presented in Fig. 19. For the top curve the rocket exhaust went entirely into the vacuum chamber, whereas the second was aligned with the scavenging inlet. As indicated in Section 2.4.3 using the chamber pressures thus obtained, it is possible to compute the mass captured by the inlet.

The fact that the rocket exhaust gases have several constituents and different individual pumping speeds apply to different gases is illustrated in Fig. 20. Three chamber conditions are shown: (1) with the chamber pumped only by the diffusion pumps, (2) with the chamber liner at 77°K, and (3) with the end panels at 20°K. In the third case, the major contribution to the total pressure comes from the H₂ which could only be pumped by the diffusion pumps.

5.2.2 Summary of Rocket Results

Plots of chamber pressure versus time for each of the inlets are shown in Fig. 21a, b, and c. From the corresponding time of rocket shutdown, the mass ratios \dot{m}_d/\dot{m}_r for each configuration are 0.55, 0.56, and 0.53. The results presented are for cold inlets and duct. The duct pressures did not rise above 5 torr for these tests, and hence it can be expected from the cold gas results that the back pressure had no effect on the mass captured by the inlets.

5.2.3 Effects of Cooling on Hot Gas Results

The results of cooling the inlets and ducts had no significant effects on the operation of the inlets. Data scatter was rather bad on some of the tests, and it is expected that results similar to the cold gas runs were experienced; that is, the inlets removed the 77°K condensables from the flow.

5.2.4 Results of Flow Reversal Results

The ejector described in Section 3.3.3 was used to prevent reverse flow into the chamber. The results of a typical test using this system are shown in Fig. 22. After rocket shutdown, the chamber pressure did not rise as when the ejector was not used indicating effective elimination of backflow into the chamber.

SECTION VI DISCUSSION

6.1 COMPARISON OF HOT AND COLD GAS RESULTS

Comparison of the results from the hot gas runs (Section 5.2.2) and those obtained with cold gas (Section 5.1.2) shows reasonably good agreement. The percentages of gas flow captured in the cold runs for low back pressure conditions ($P_d \rightarrow 0$) were 0.75, 0.50, and 0.50 for the three inlets, whereas the hot gas runs gave corresponding values

of 0.55, 0.56, and 0.53. Since one of the major concerns pertained to the applicability of cold gas results when using hot rocket firings, this correlation was significant. It should be remembered that the inlets had been located at different positions with respect to the nozzle exits to capture equivalent percentages of the flow. Thus, the correspondence between cold and hot results further substantiates the computation of the plume geometries.

6.2 COMPARISON OF ESTIMATED PERFORMANCE WITH EXPERIMENT

Analysis had been performed which indicated that the inlets should capture near 94 percent of the rocket flow with a maximum duct pressure of 10 to 15 mm Hg. The maximum calculated pressure recoveries compared favorably with experiment; however, the amount captured by the inlets was considerably below the anticipated amounts except for the straight duct with a low back pressure (Fig. 17). In an attempt to find the cause for this discrepancy, a more detailed review of the flow approaching these inlets was considered. Two points of possible significance are apparent. First, a reassessment of the viscous effects indicates that boundary-layer buildup inside the lip of the inlets may be much more than was originally estimated. Second, and to some extent combined with the first effect, the flow near the outer edge of the captured gas is at the highest Mach number and lowest static pressure. When the gas passes through the shock system formed in the inlet, the center portion of the flow can be at a significantly higher pressure than the flow near the outer edge. This pressure nonuniformity together with the large boundary-layer leads to a situation (Fig. 23) where the numbers shown are the relative distribution of pressures that would exist if the gas had passed through a normal shock. Of course, the flow adjusts to equalize the pressures; thus, considerable backflow and spillage results. It had been anticipated that mixing would help to distribute the energy across the duct, but the rapid buildup of boundary layer counteracts this effect.

The low duct pressure data which were taken with the straight duct indicated that its spillage could be low at low back pressure; however, for the two centerbody inlets the flow area at the throats was not sufficient to allow the total flow to pass through with such a large boundary layer in existence. Thus, the inlets choked and only about 50 to 60 percent of the design flow was able to pass through the inlets even with the back pressure at a low value. For satisfactory operation of a centerbody-type diffuser in this kind of flow field (i.e., low pressure, high Mach number) a better design might be achieved with a small angle cone, less than 20-deg half-angle, and with twice the throat area used

in these tests. In addition, if the scavenging system can possibly be designed to accept less than 90 percent of the rocket mass flow, this will materially help the nonuniform Mach number situation and the extremely high Mach number at the lip of the diffuser.

Although no data were obtained using the straight duct inlet when it was set to capture less than the design flow, it is believed that improvement in its performance could also be achieved by having it capture less of the nozzle flow, thus reducing the entering nonuniformities. Of course, in any given installation one does not have the freedom to arbitrarily choose the amount bypassed around the inlet since this amount must balance with the test requirements and the space chamber pumping speeds.

As a comparison with previous ejector-diffuser work, the pressure ratio p_r/p_c in the current studies is about 5×10^6 , whereas in previous work (Ref. 8) the maximum ratios obtained were about 1.6×10^4 and in Ref. 9 about 2.5×10^4 .

6.3 INLET COOLING AND REVERSE FLOW INHIBITORS

The effects of cooling the inlets and duct were rather minor except for the fact that cooling the inlets to LN_2 temperature effectively removes the H_2O and CO_2 from the flow, thus reducing the external pumping load. Since LN_2 cooling is relatively cheap and easily accomplished, it is probably profitable to consider cooled inlets for most tests.

The use of the CO_2 ejector effectively prevented the gas in the scavenging duct from flowing back into the chamber when the rocket shut down. Without the reverse flow ejector, almost 75 percent of the rocket mass flow went back into the chamber. Of course any quick-acting valve or reverse flow restriction will function to give the same effect. If the scavenging pump can be located such that the volume of the scavenging duct is small, then the amount of gas contained in the duct is small and no separate reverse flow device is needed.

6.4 SYSTEM UTILIZATION IN A TYPICAL SPACE CHAMBER

The utilization of the system can be seen in the following example: It is desired to test a rocket of approximately 50-lb thrust while maintaining a pressure below 1×10^{-4} torr within the space simulation chamber. The rocket mass flow is 0.17 lb/sec with a chamber pressure of 500 psia, and firings of 5-sec duration are desired. The exhaust

gas products by weight are 1×10^4 torr ℓ /sec of 77°K condensable (H_2O , CO_2 , etc.), 5×10^3 torr ℓ /sec of 20°K condensable (N_2 , O_2 , etc.), and 5×10^3 torr ℓ /sec of hydrogen all at 77°K.

The approximate size of the pumps available in the chamber are: (1) the 10^4 ℓ /sec roughing pumps which can pump the gas in the scavenging system, (2) the 10^8 ℓ /sec LN_2 surfaces for 77°K condensables, (3) the 10^7 ℓ /sec GHe surfaces for 20°K condensables, and (4) 10^6 ℓ /sec diffusion pumps for hydrogen.

Using the total mass flow and scavenging pumping system to estimate minimum pressure which can be maintained in the scavenging duct, a pressure of 5 mm is obtained. The p_d/p_r ratio is then 2×10^{-4} which is about the maximum which should be used if the inlet is allowed to spill about 10 percent. The next step is to see if the chamber can maintain the desired pressure with 10 percent of the mass spilling into the chamber. At the end of the 5-sec run, the partial pressures contributed by the various components would be: 1×10^{-5} torr by 77°K condensables, 5×10^{-5} torr by 20°K condensables, and 1×10^{-3} torr by the hydrogen. The sum of the 77 and 20°K condensables is 6×10^{-5} torr; however, the hydrogen partial pressure is one decade too high and either the spillage flow must be cut to less than 1 percent or the hydrogen pumping speed must be increased. Since the current data show that we cannot hope to reduce the spillage to 1 percent and operate successfully, something must be done to improve the pumping capacity in the scavenging system to handle the flow at lower pressure or to improve the chamber pumping capacity for hydrogen.

In Refs. 1 and 12, it can be seen that the chamber pumping speed for hydrogen can be increased by depositing titanium on the 77°K surfaces so that they will cryosorb hydrogen. Using data in these references, the amount of 77°K surfaces which must be covered with a titanium film is computed to be 2000 ft^2 or about 20 percent of the chamber LN_2 -cooled surfaces. This gives a total H_2 pumping speed of 10^7 ℓ /sec and a corresponding partial pressure of 5×10^{-5} torr. With this H_2 pumping capacity, the chamber total pressure is 1.1×10^{-4} torr or satisfactorily close to the 1×10^{-4} torr pressure desired.

The installation would be about as shown in Fig. 24 with a 24-in. inlet and duct located near the rocket. Since the duct volume is relatively large, some type of reverse flow device is required to prevent backflow into the chamber. A simple venetian blind-type valve operated by the rocket exhaust has been designed which should serve as an effective reverse flow device.

SECTION VII CONCLUSIONS

The following conclusions can be drawn from the work reported herein:

1. The plume computational technique provides a suitable technique for computing plume properties.
2. The pressure recoveries for the inlets were about as estimated with little difference between the inlets tested.
3. The mass captured by the inlets was not as high as anticipated. Redesign and operation with 10 percent or more spillage should improve performance.
4. A reverse flow inhibitor can be successfully applied to minimize backflow into the chamber.
5. Proper application of cryosorption and scavenging techniques to rocket exhaust products can extend the range of systems tests in space environmental chambers.

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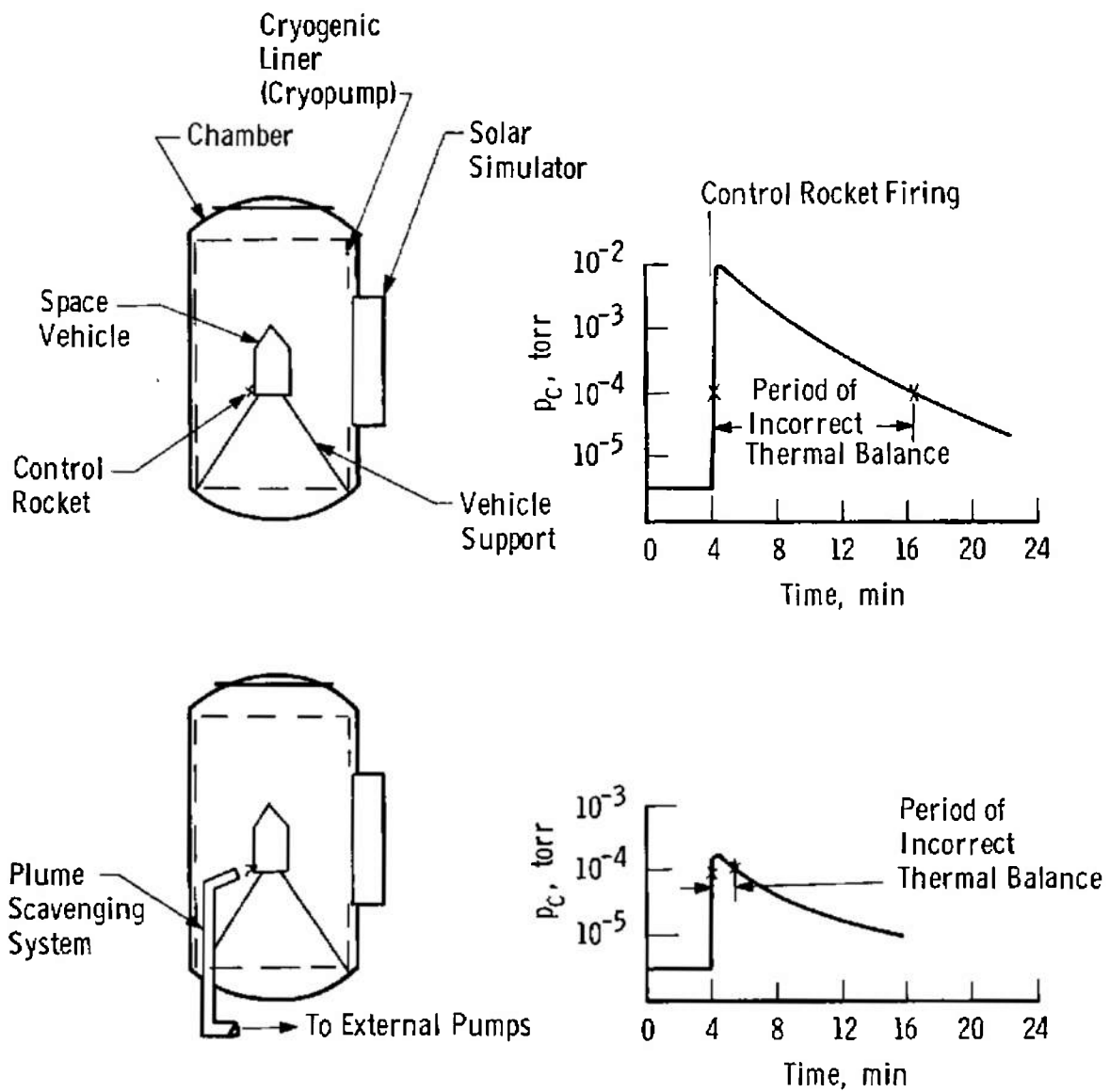


Fig. 1 Test Schematic Showing the Use of a Scavenging System

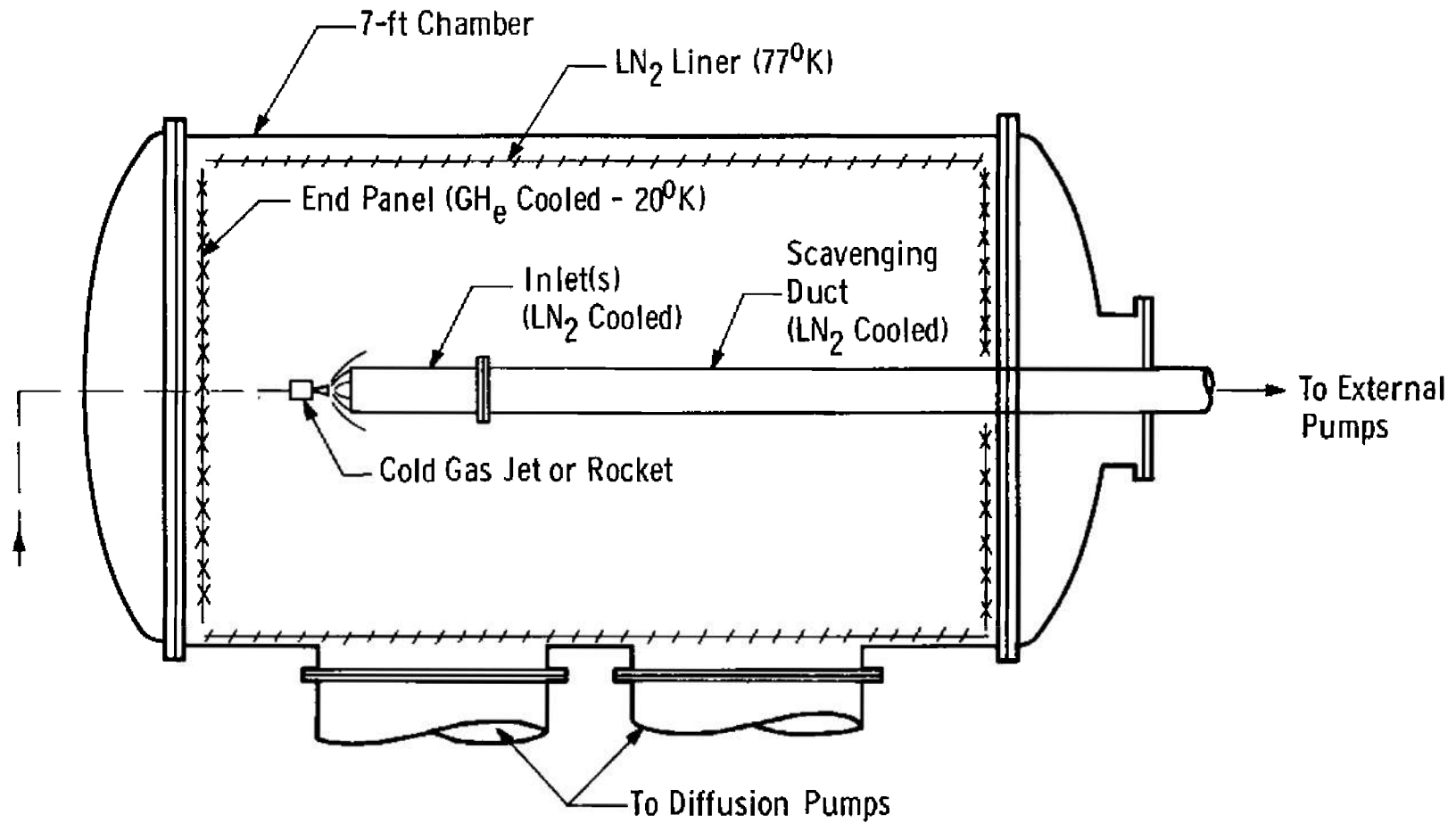


Fig. 2 ARC (7V) Installation Schematic

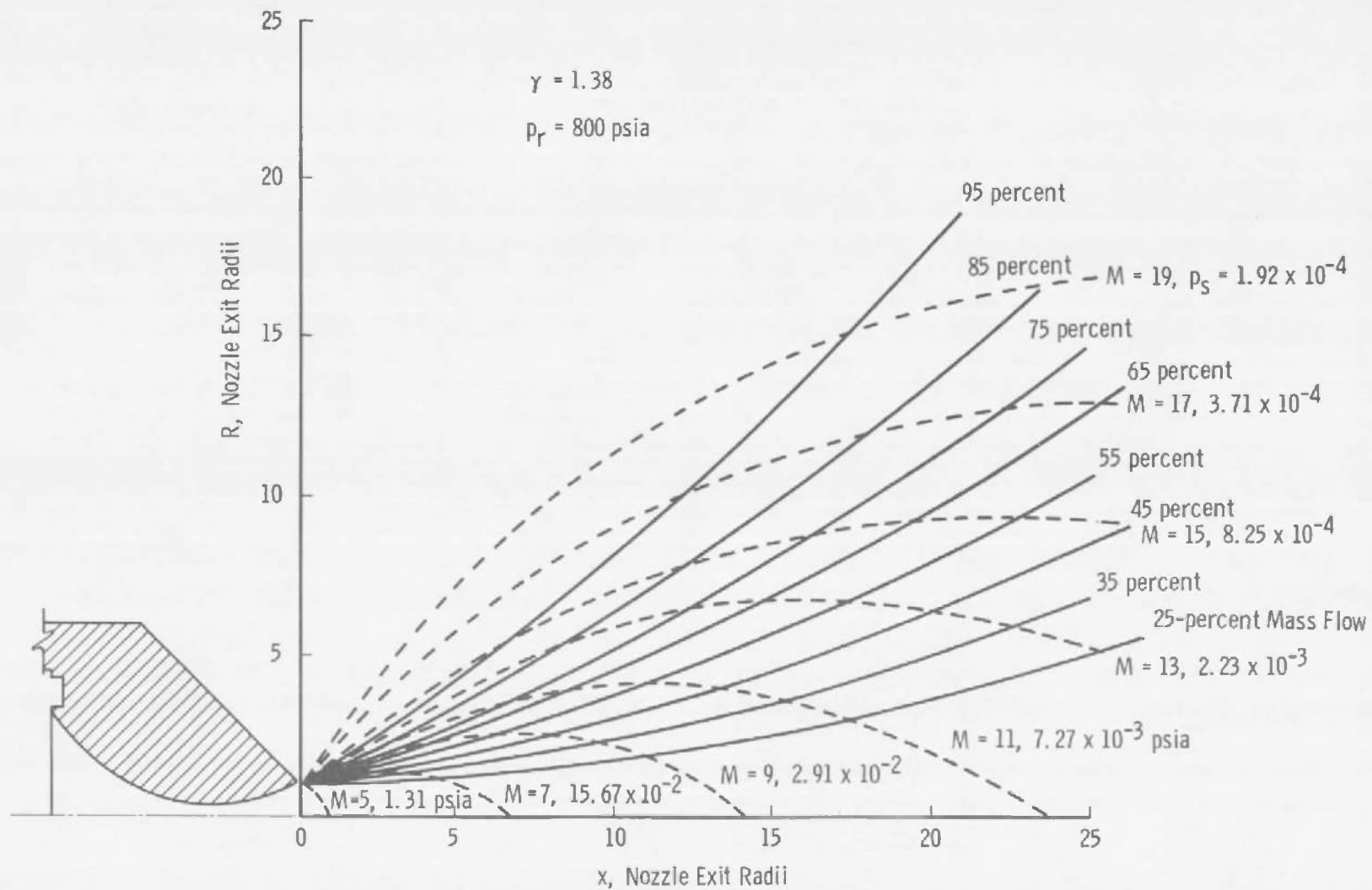


Fig. 3 Cold Gas Plume Characteristics

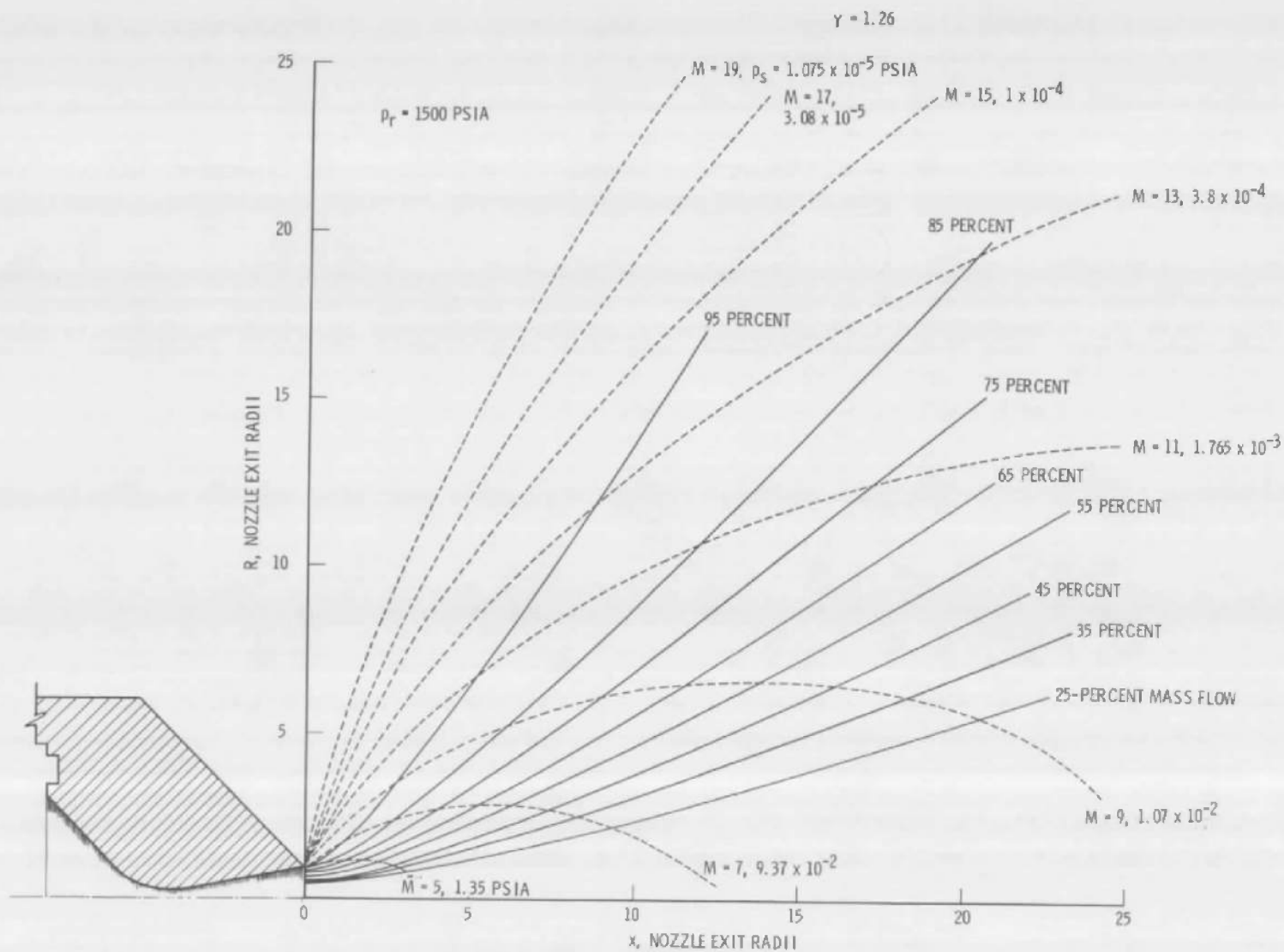


Fig. 4 Hot Gas Plume Characteristics

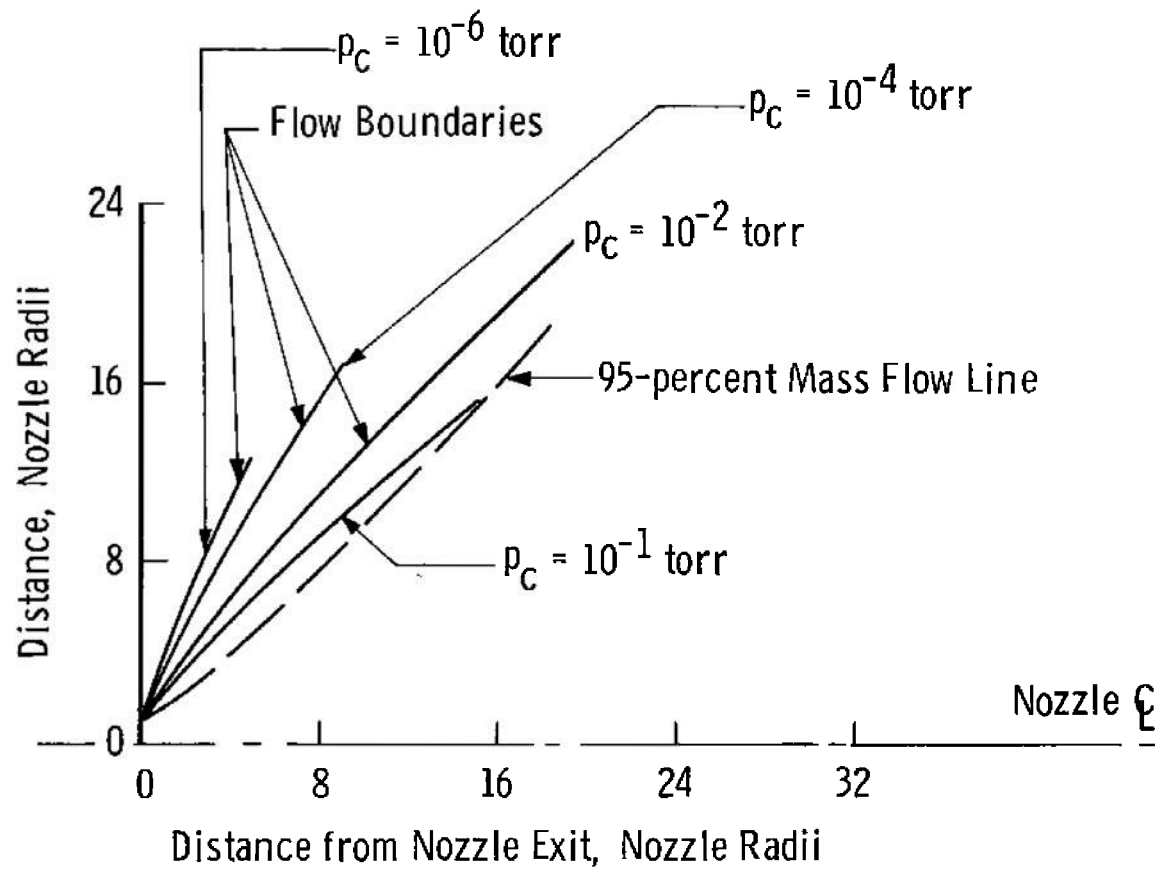


Fig. 5 Flow Boundaries with Different Chamber Pressure

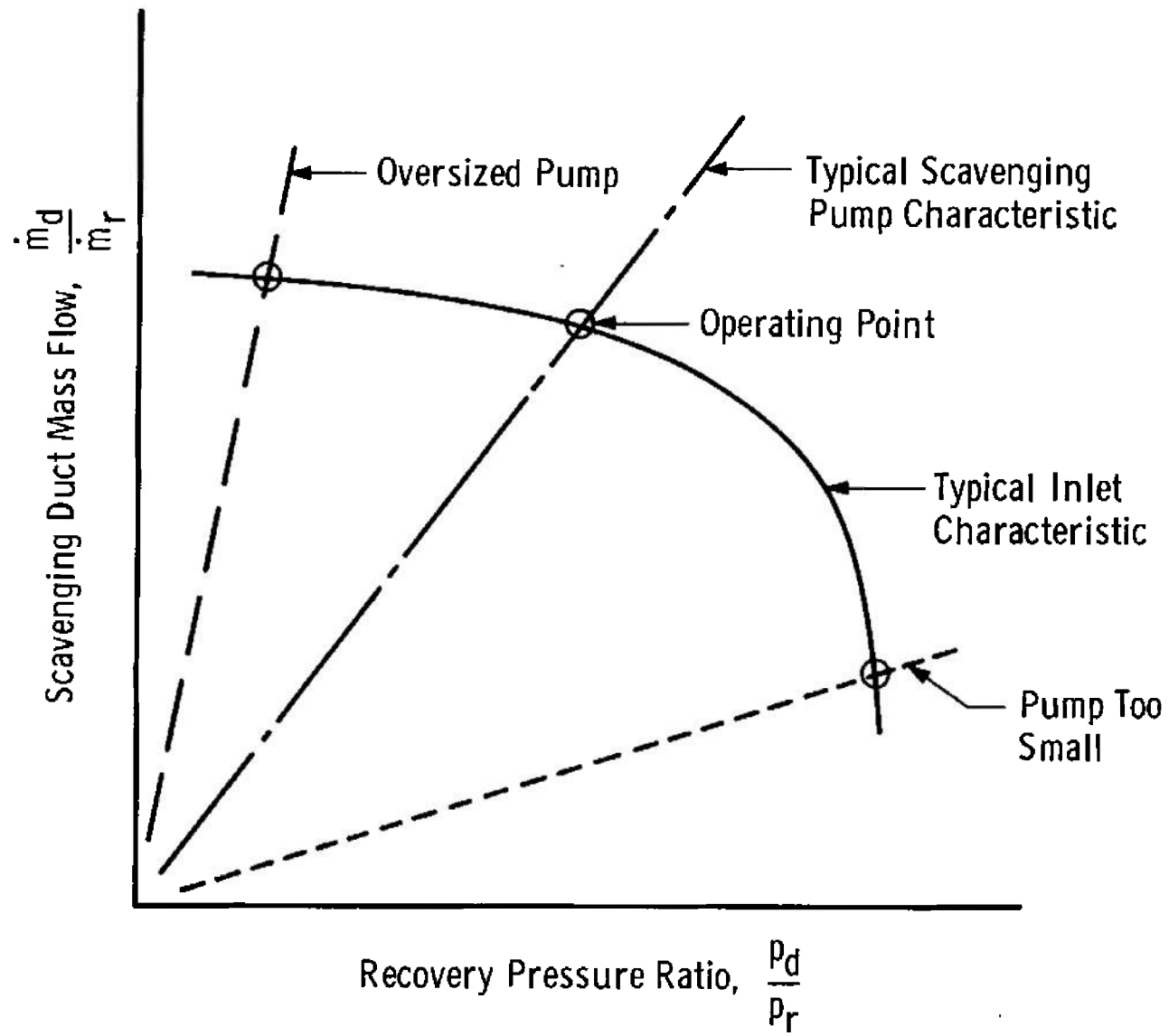


Fig. 6 Matching of Inlet and Pump Characteristics

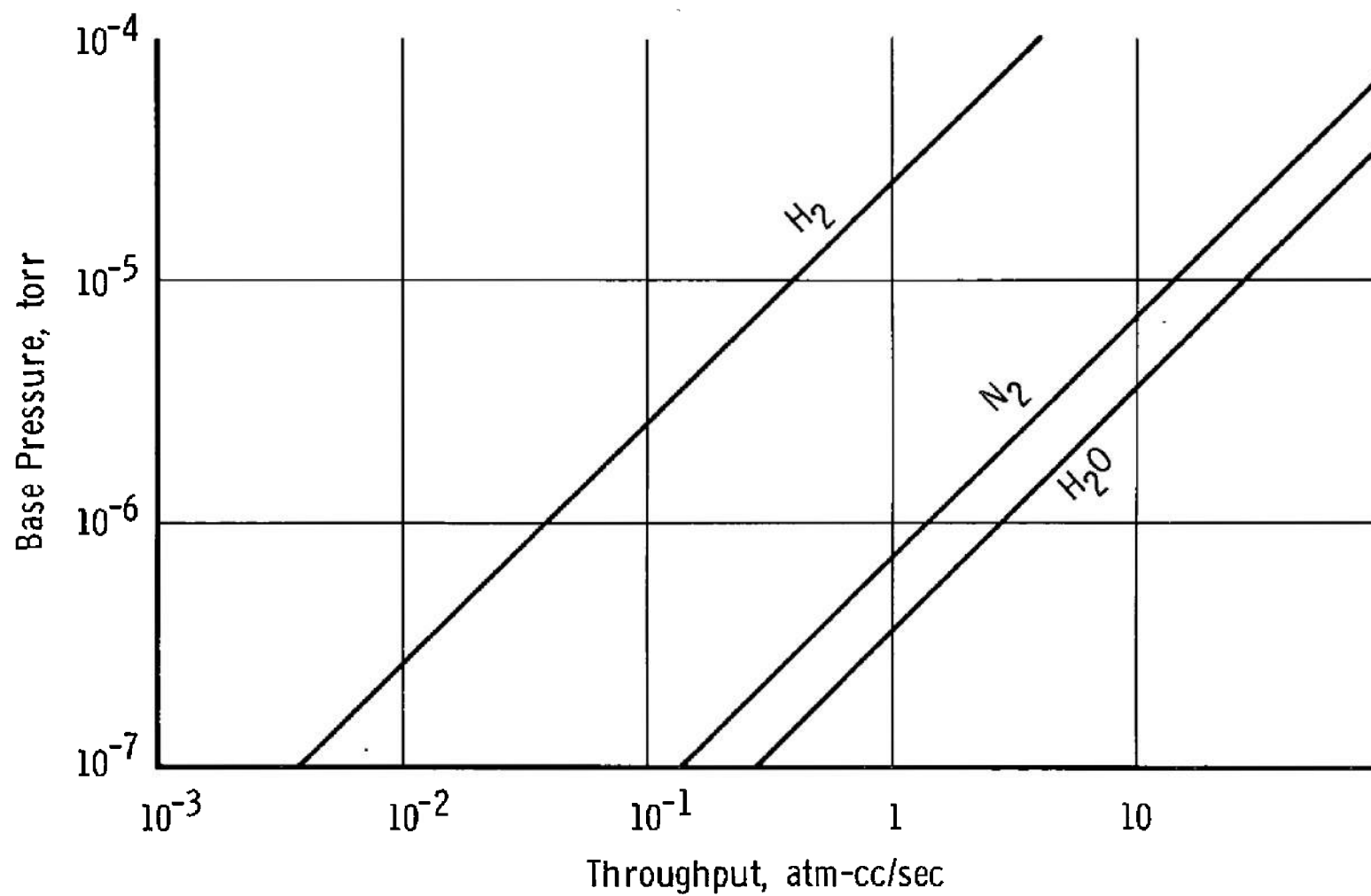


Fig. 7 ARC (7V) Pumping Speed (Basic Configuration)

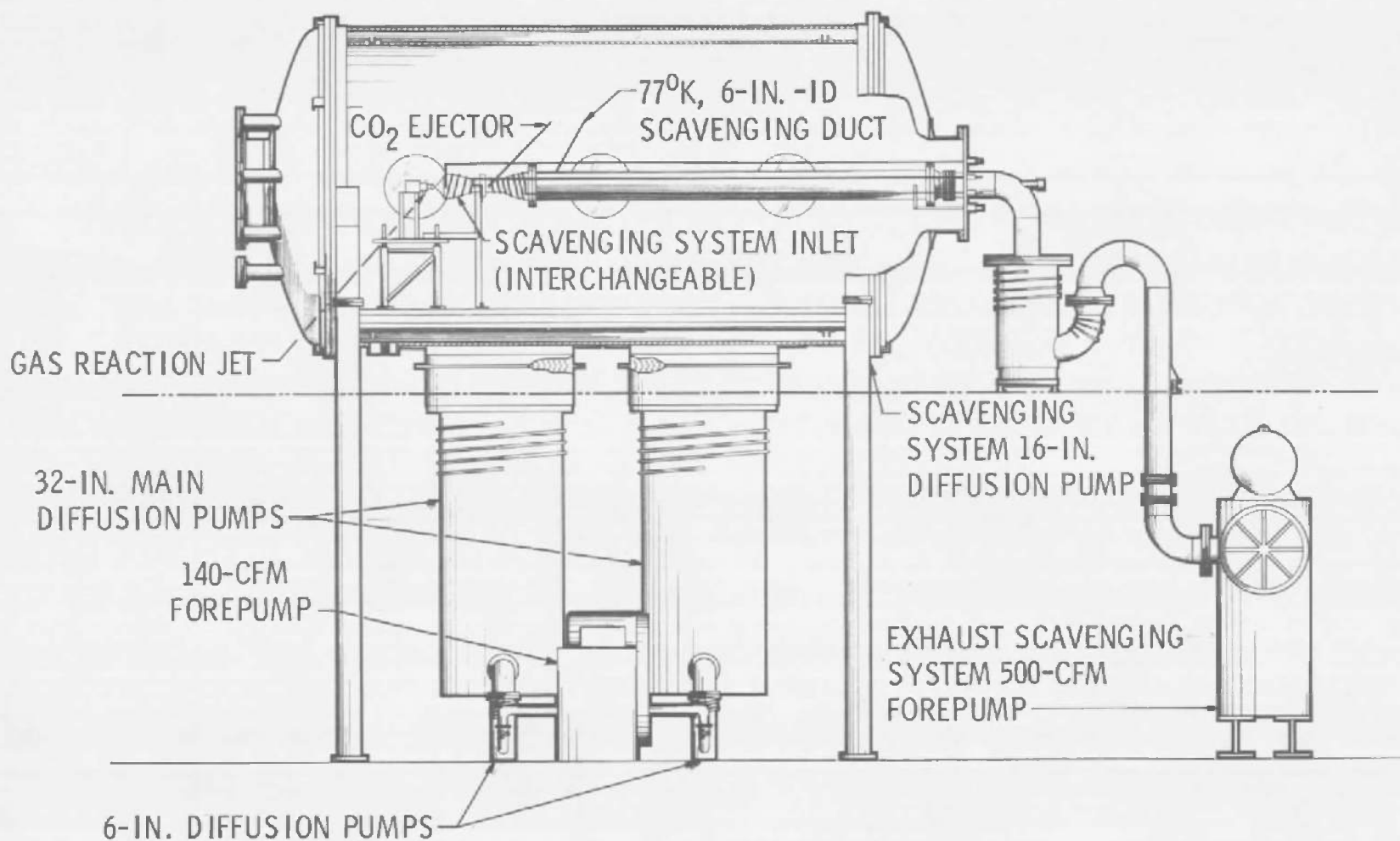
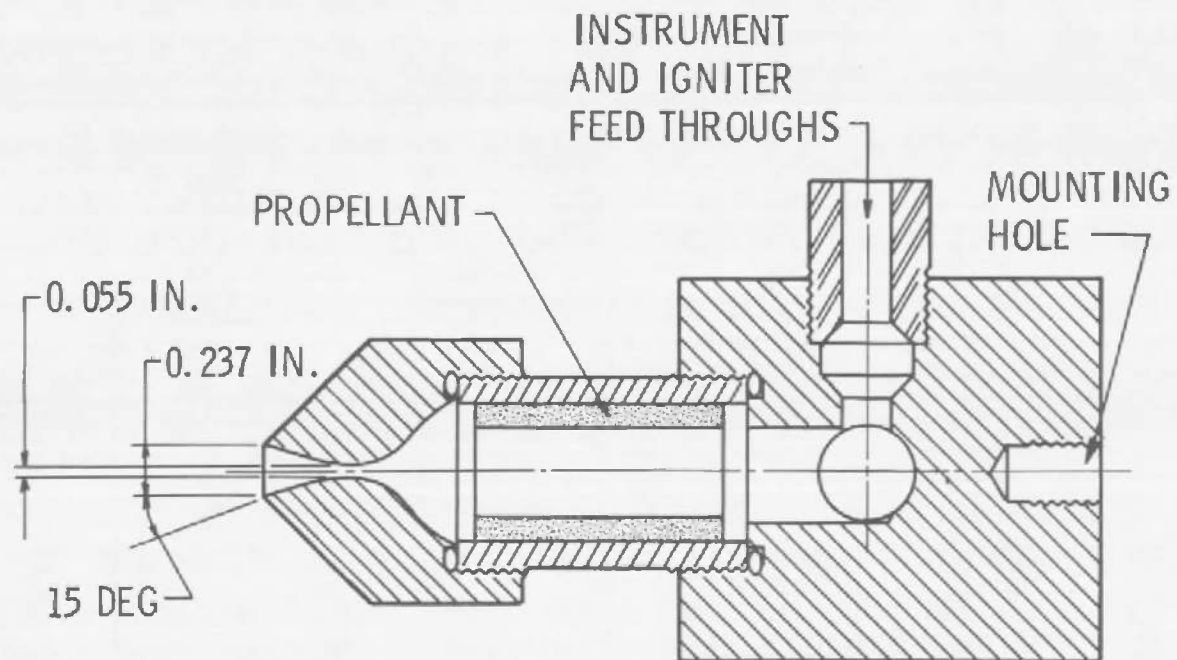


Fig. 8 ARC (7V) with Exhaust Scavenging System Installed



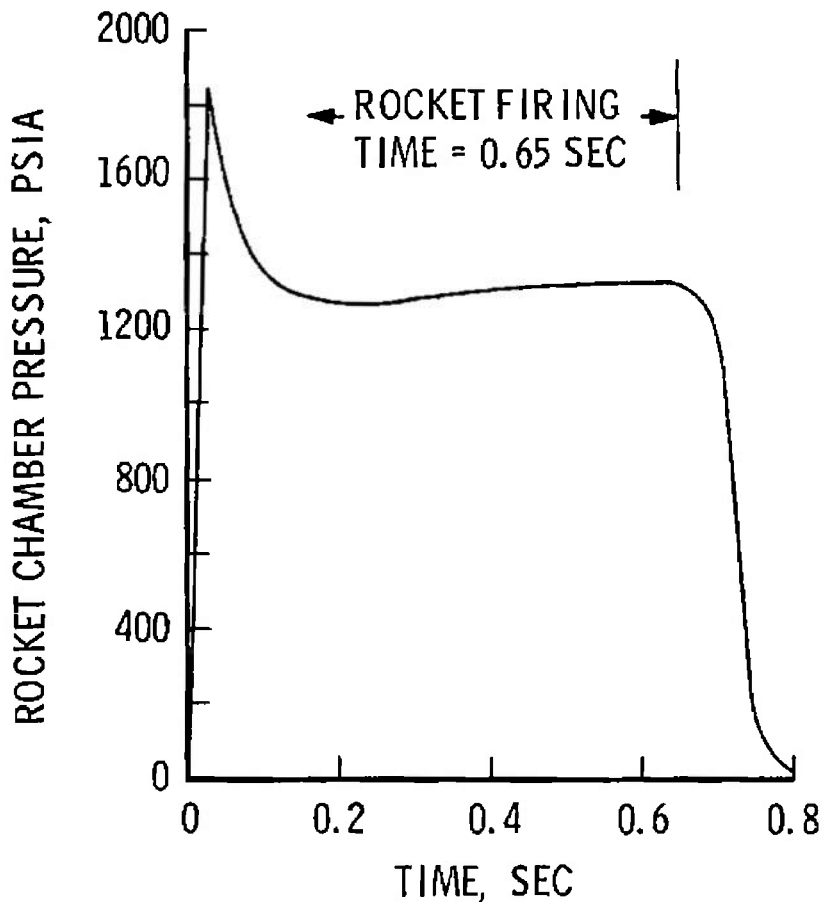
a. Assembly

Fig. 9 Gas Generator and Typical Rocket Chamber Pressure

ROCKET GAS	EXHAUST PERCENT BY WEIGHT	COMPOSITION PERCENT BY VOLUME
CO	53.80	41.7
CO ₂	14.53	7.5
H ₂	1.87	21.5
H ₂ O	13.60	16.5
N ₂	16.15	12.8

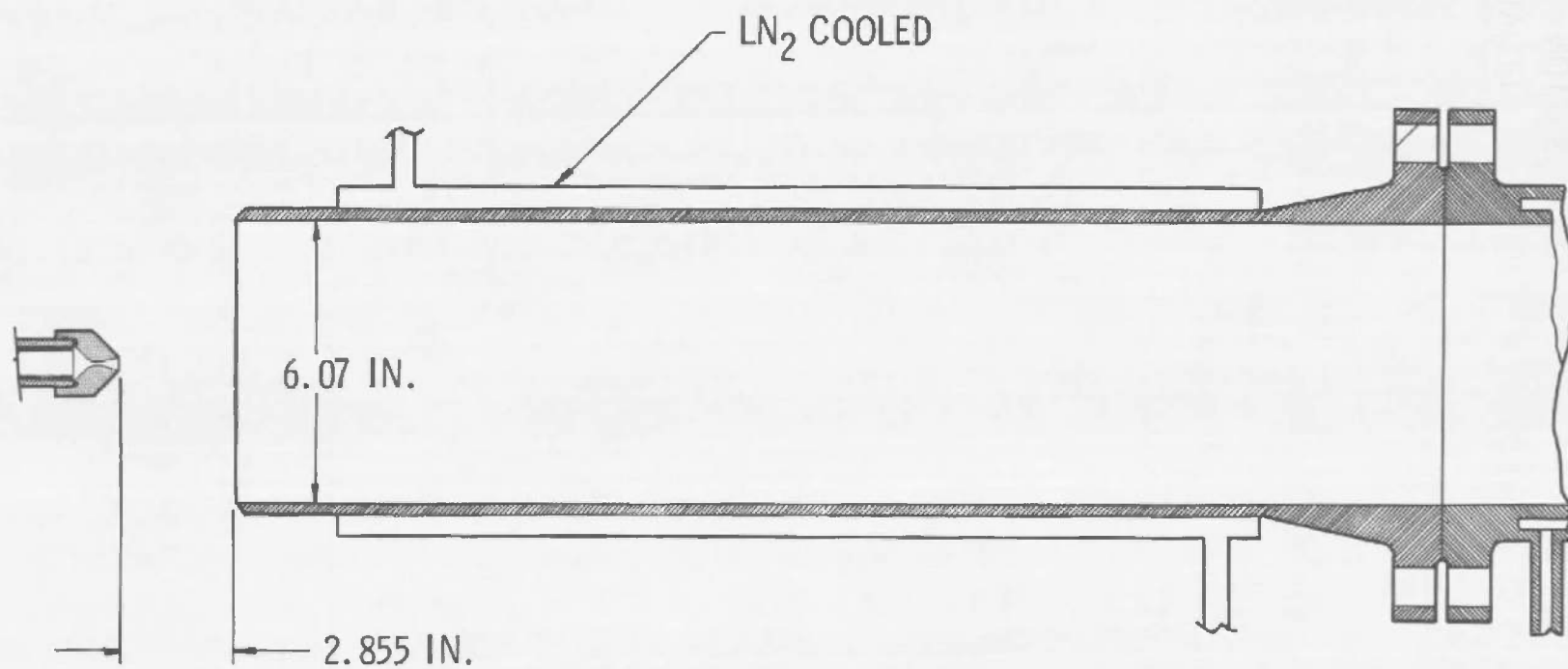
ROCKET FIRING TIME = 0.65 SEC

MASS FLOW OF ROCKET EXHAUST = 0.025 LB_M/SEC

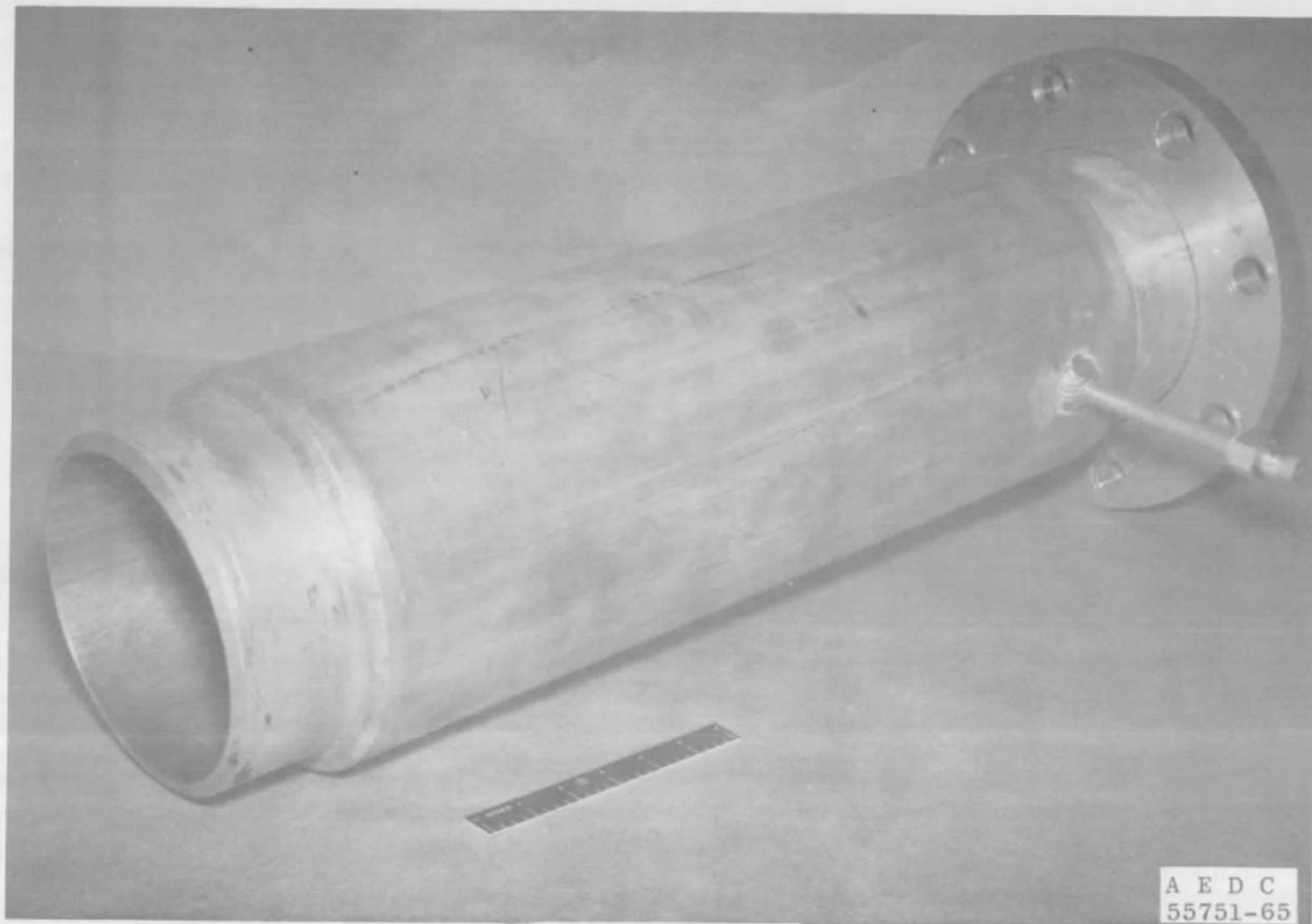


b. Chamber Pressure

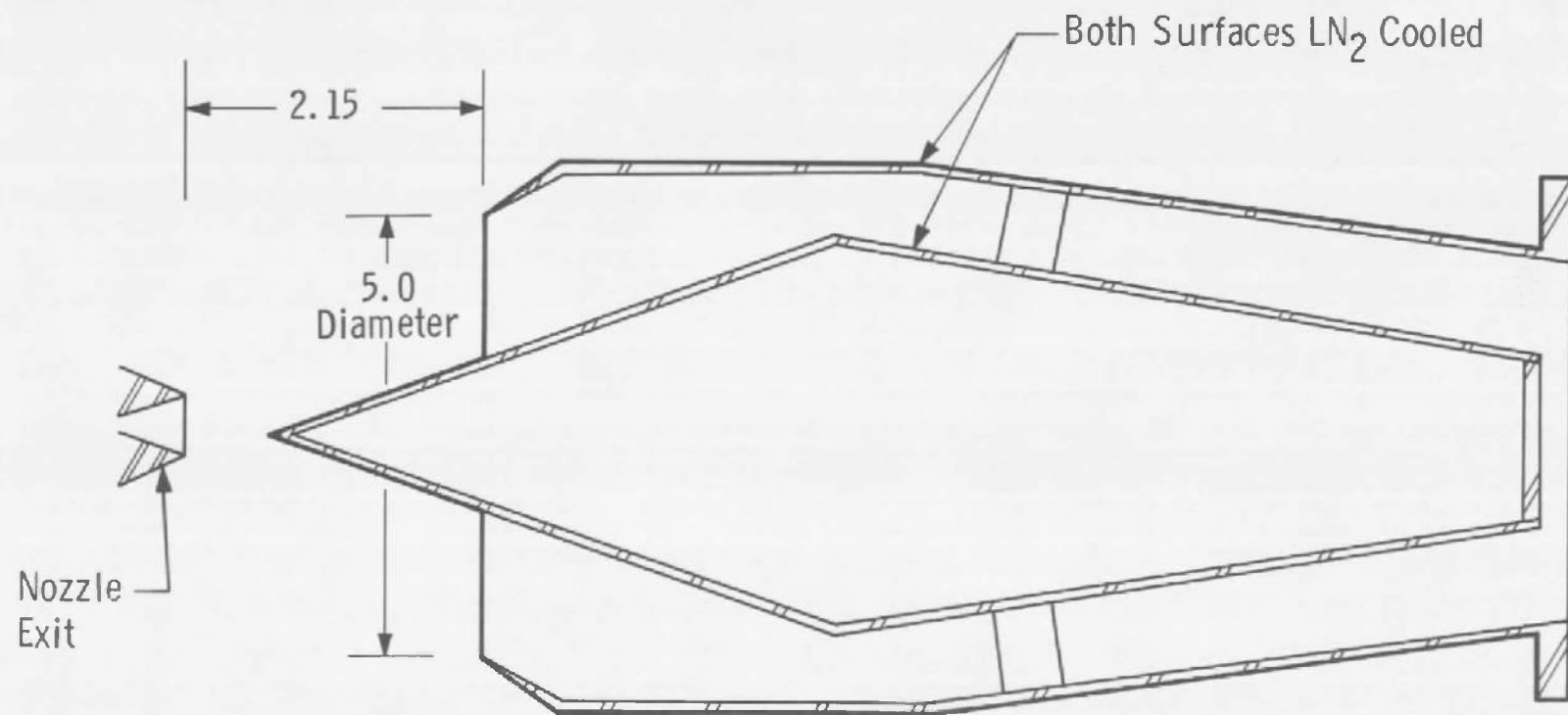
Fig. 9 Concluded



a. Schematic
Fig. 10 Configuration A

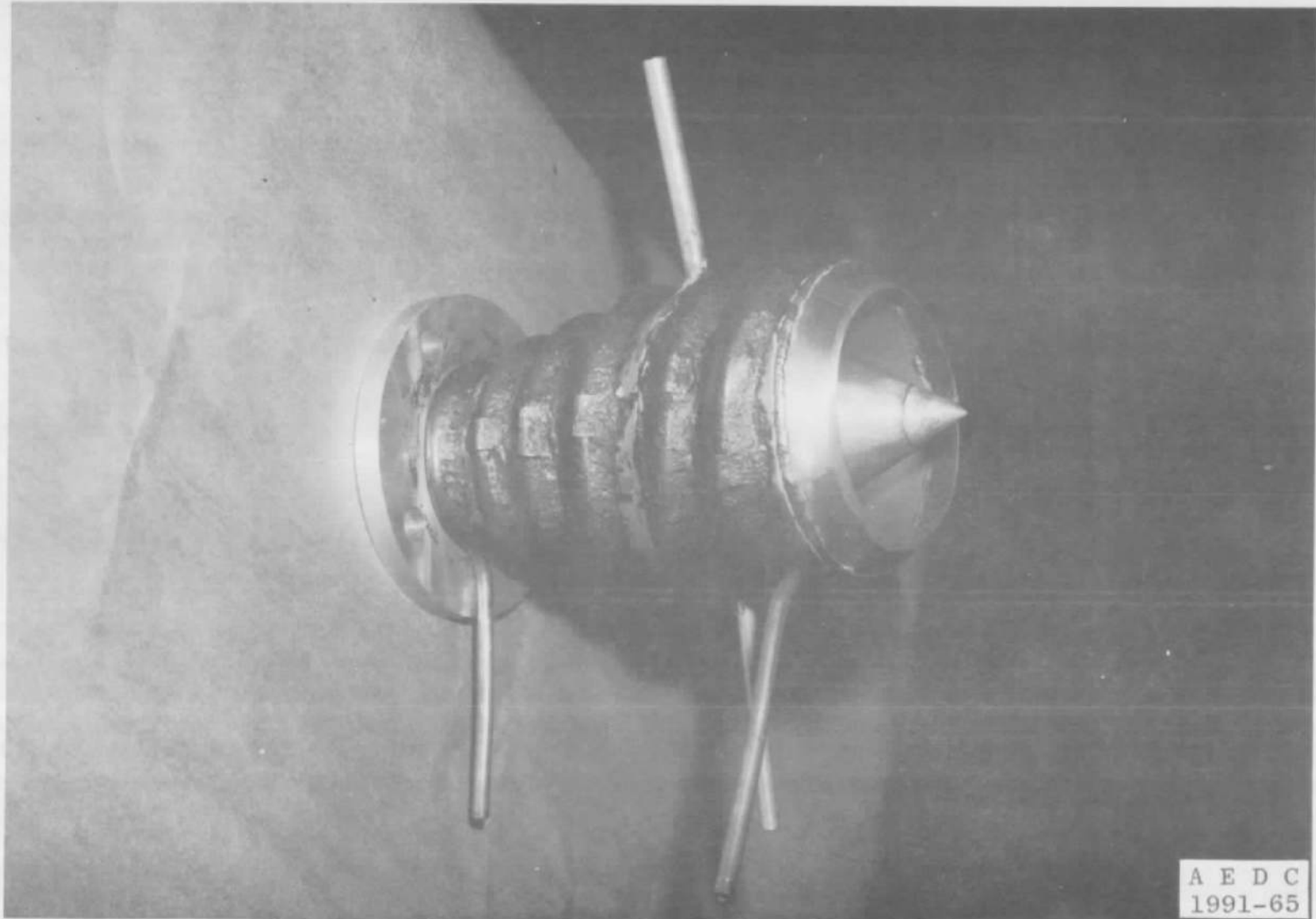


b. Photograph
Fig. 10 Concluded



All Dimensions in Inches

a. Schematic
Fig. 11 Configuration B



b. Photograph
Fig. 11 Concluded

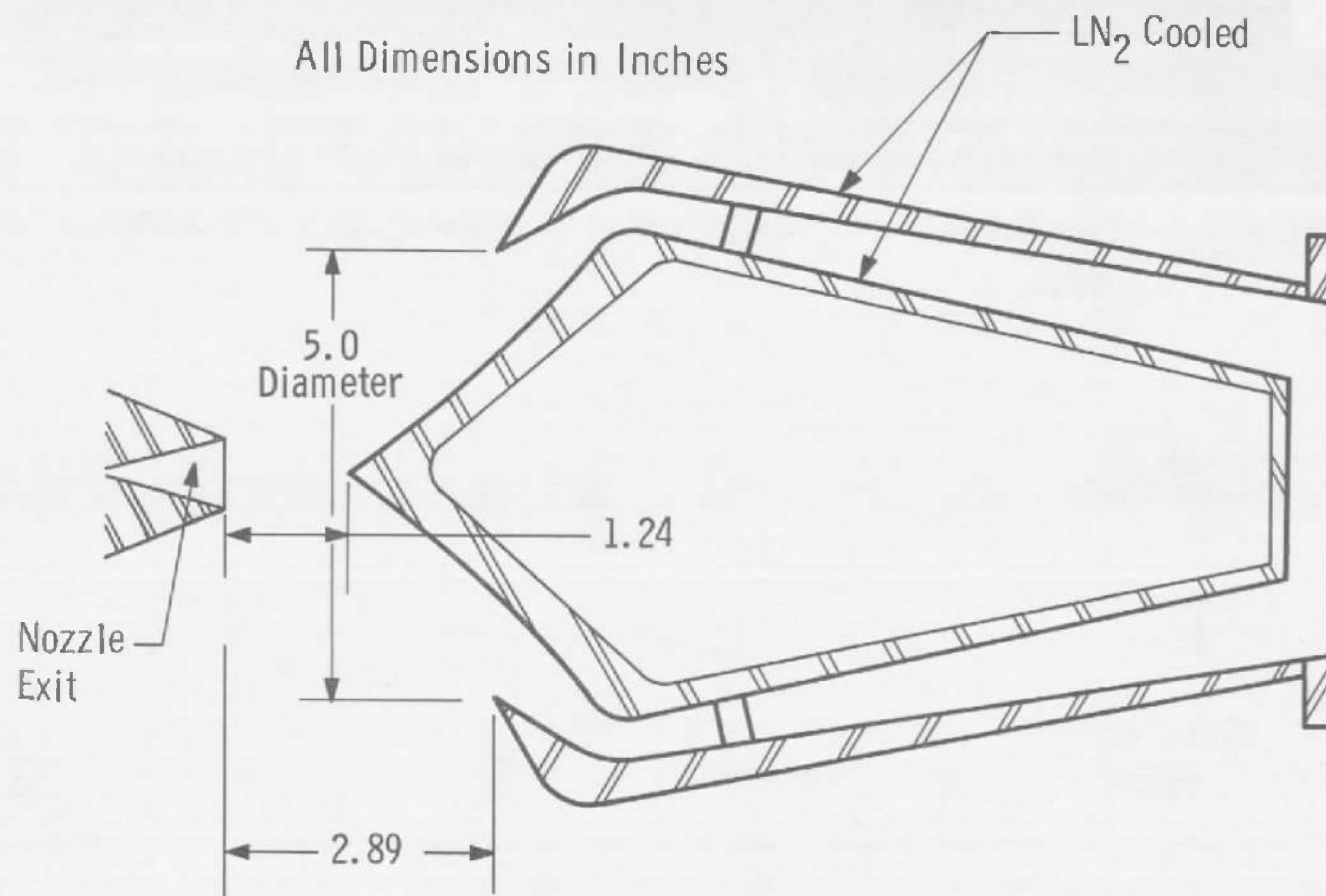


Fig. 12 Configuration C

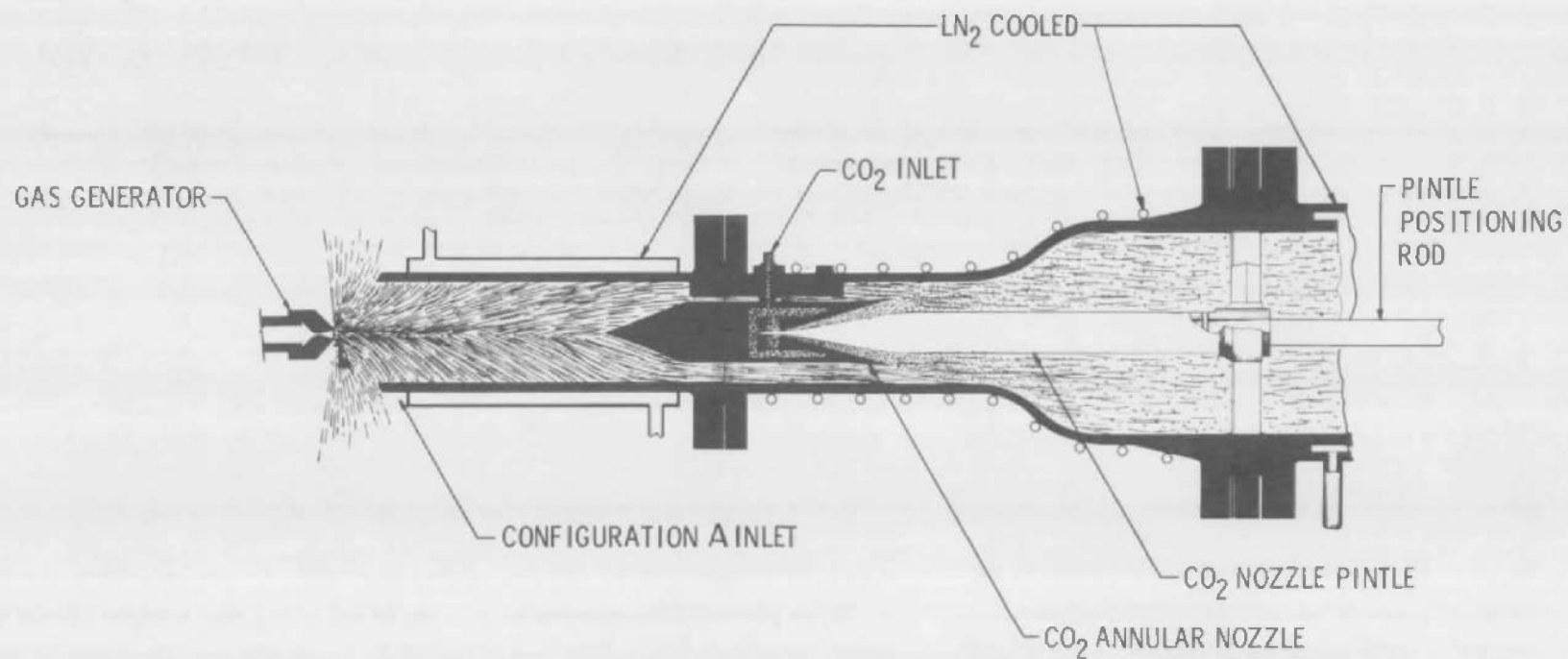


Fig. 13 Schematic of Exhaust Gas Scavenging System Showing CO₂-Driven Reverse Flow Inhibitor

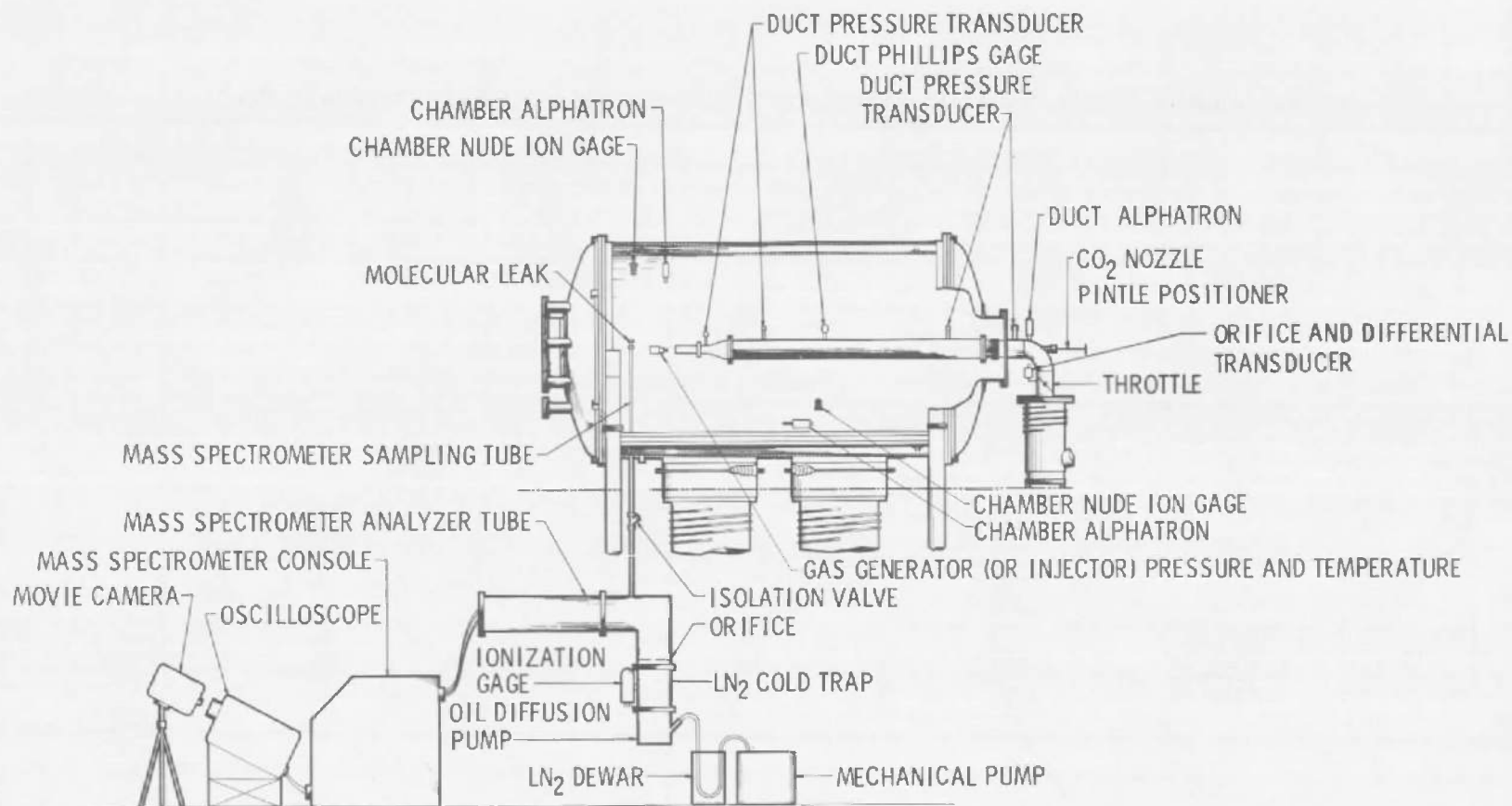


Fig. 14 Instrumentation Schematic

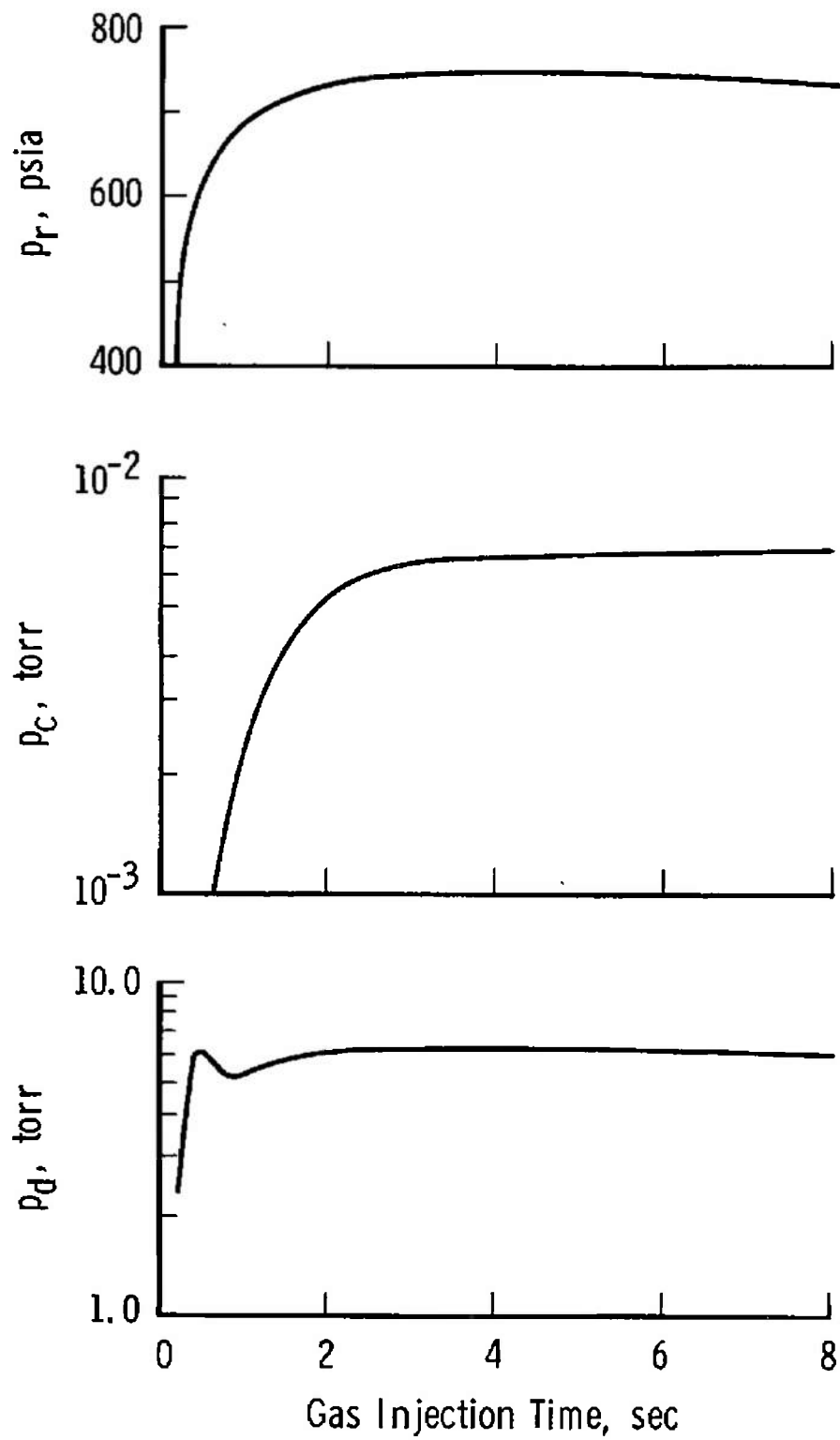


Fig. 15 Typical Cold Gas Run

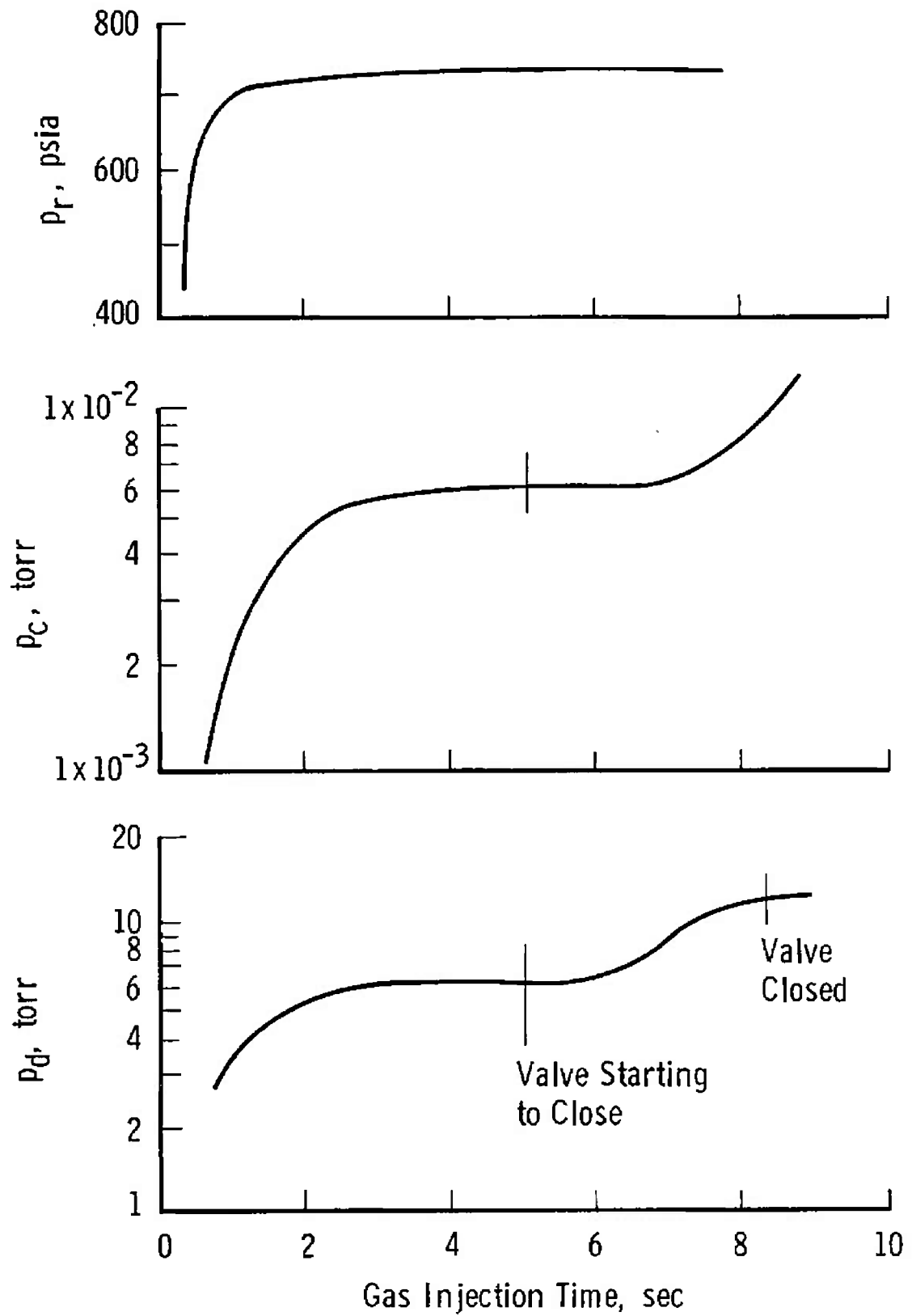


Fig. 16 Cold Gas Run with Throttle Valve Closing

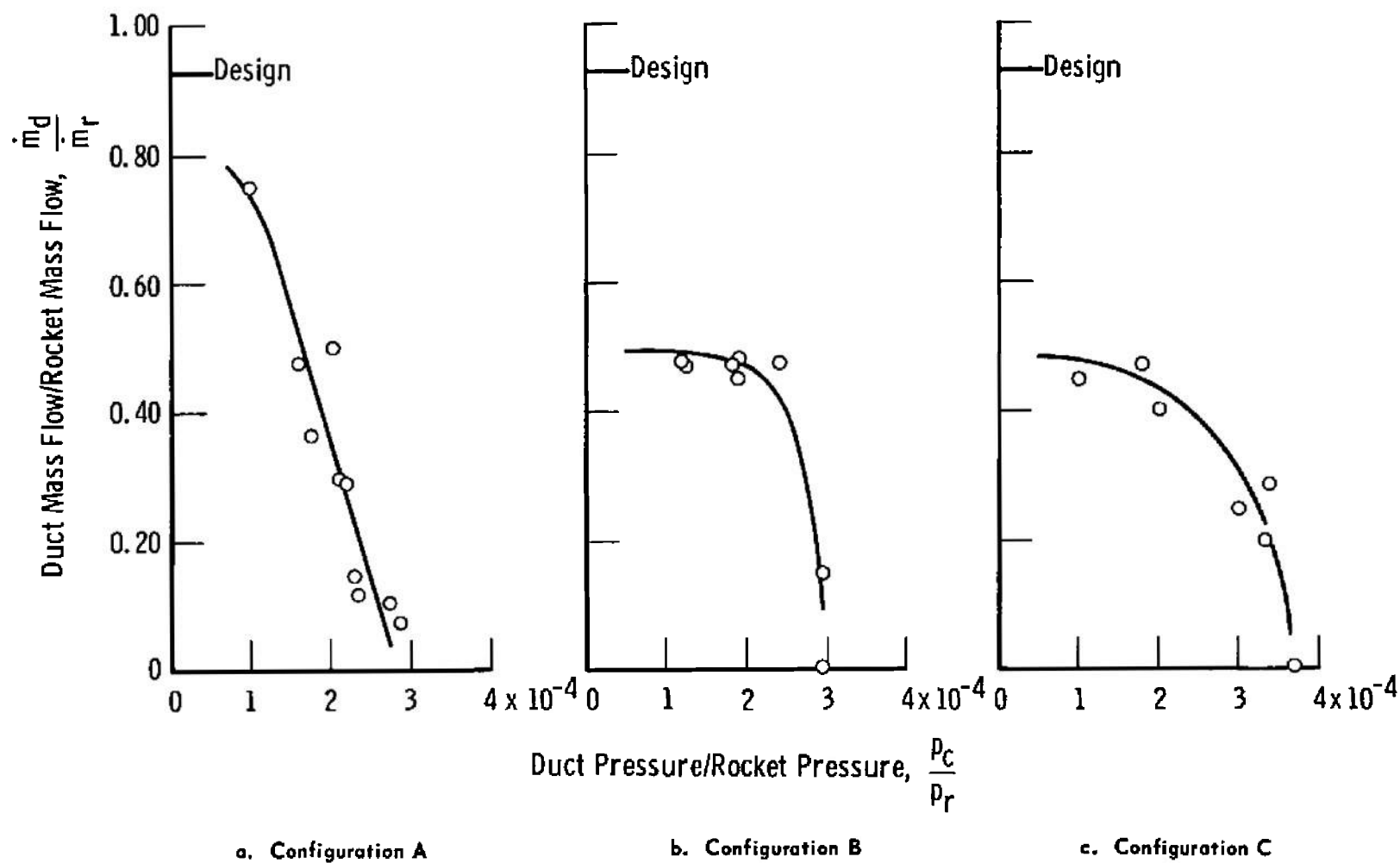


Fig. 17 Cold Gas Recovery Data

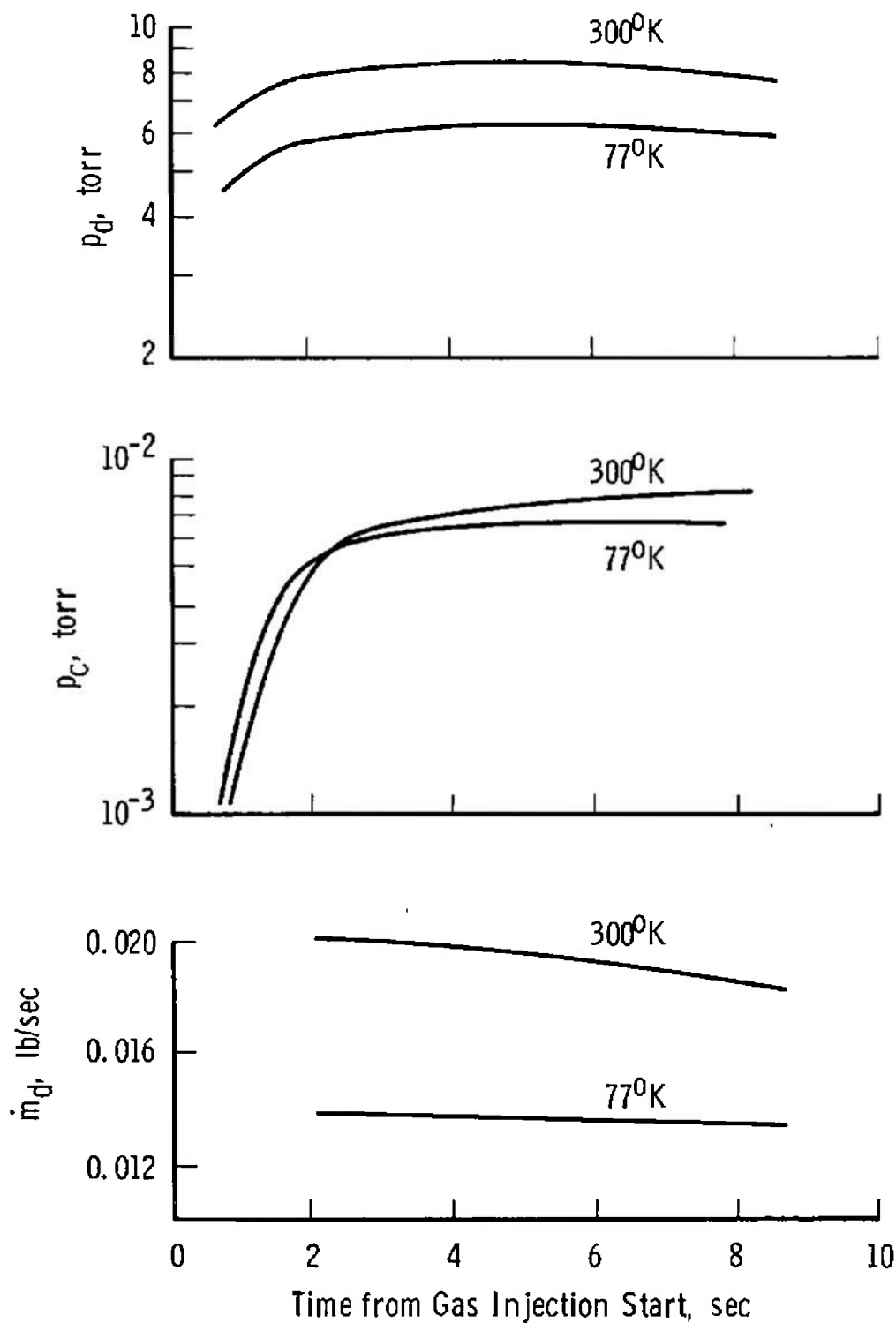


Fig. 18 Effect of Duct Cooling (Cold Gas)

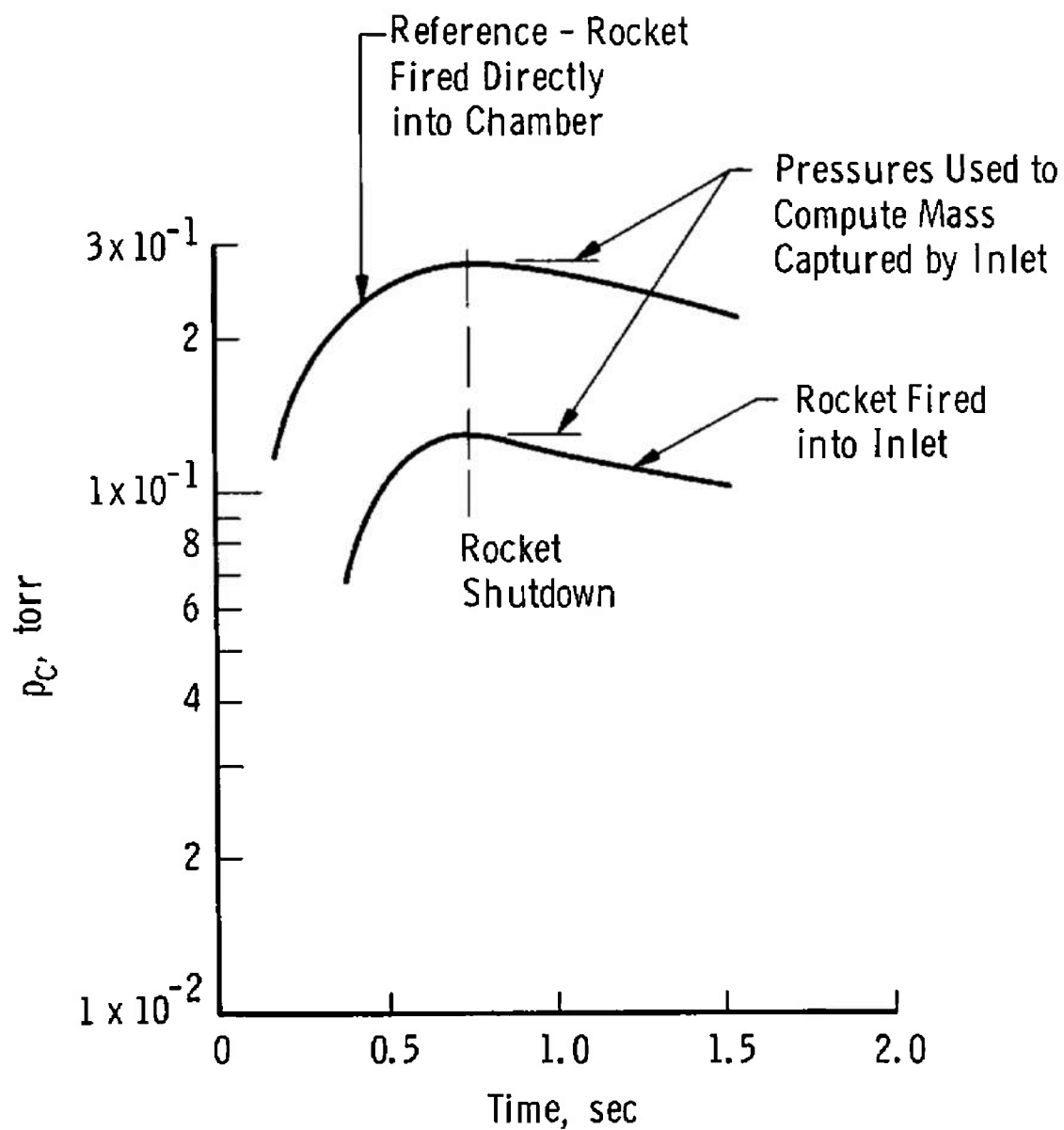


Fig. 19 Typical Hot Gas Run

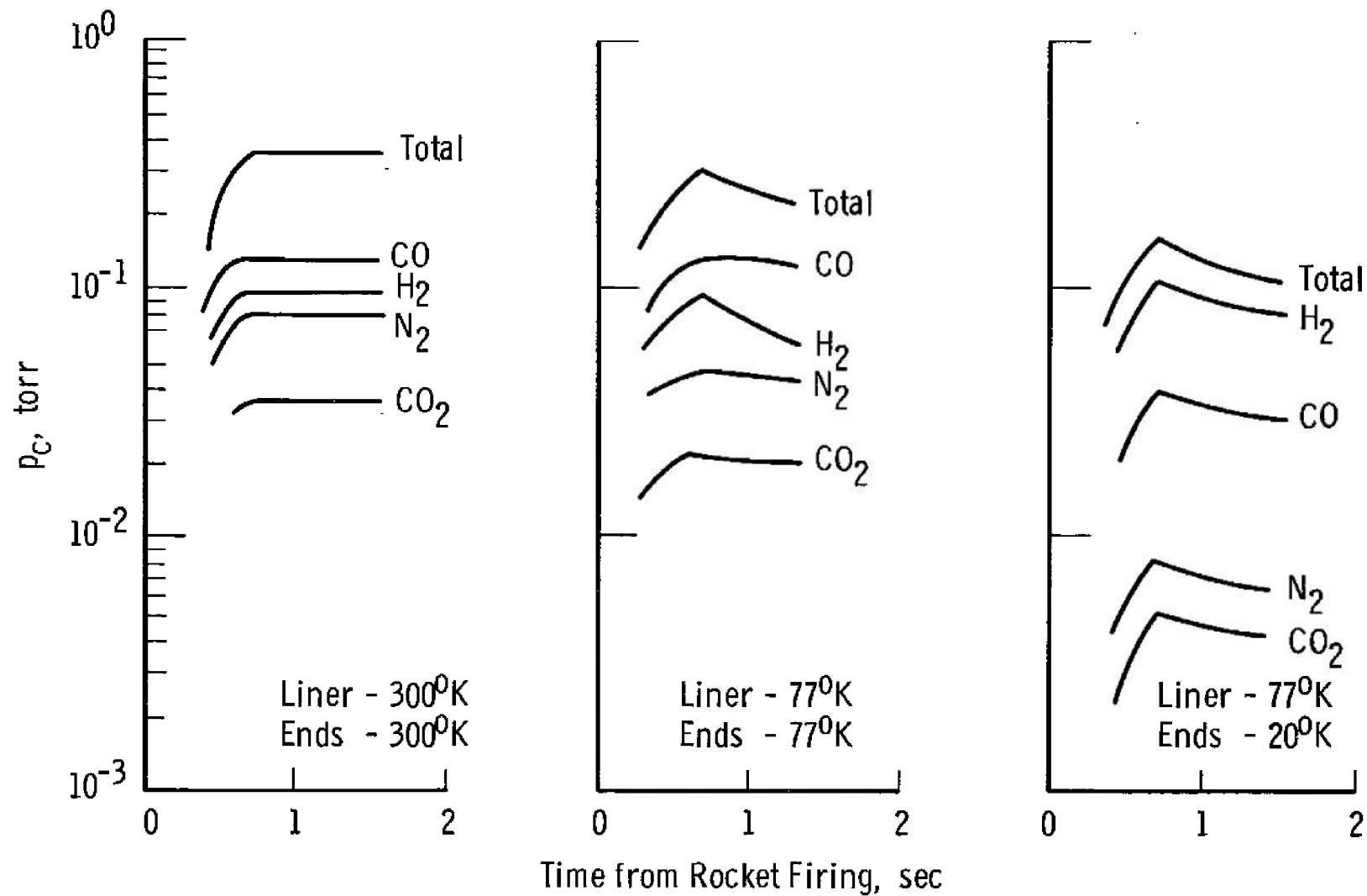
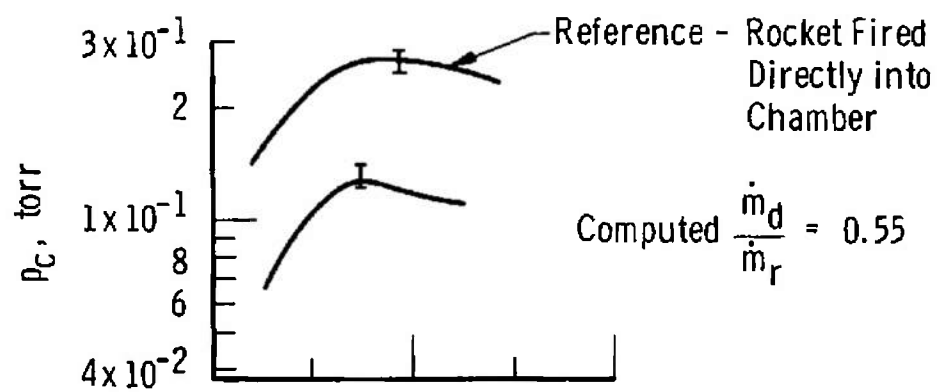
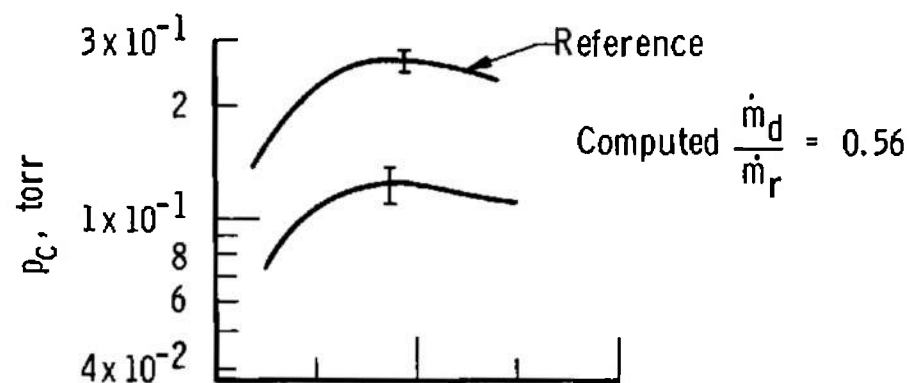


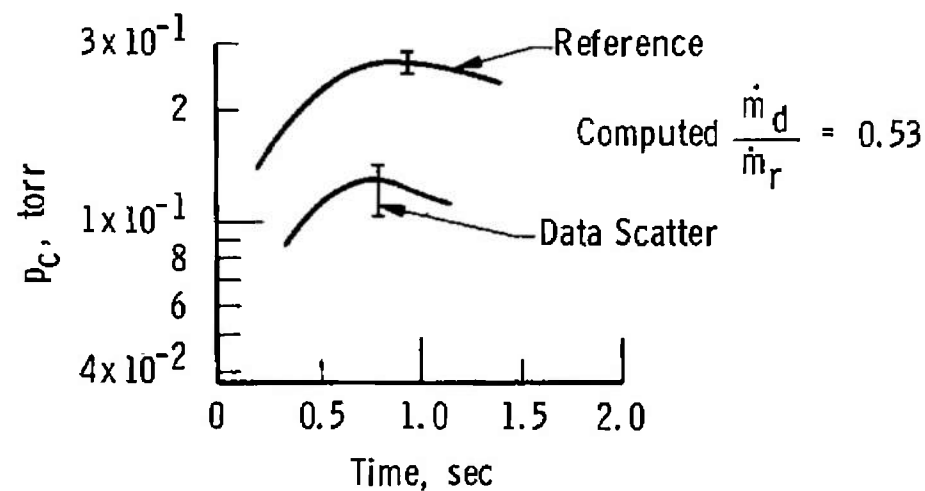
Fig. 20 Chamber Partial Pressure with Different Cold Wall Conditions



a. Configuration A



b. Configuration B



c. Configuration C

Fig. 21 Hot Gas Results

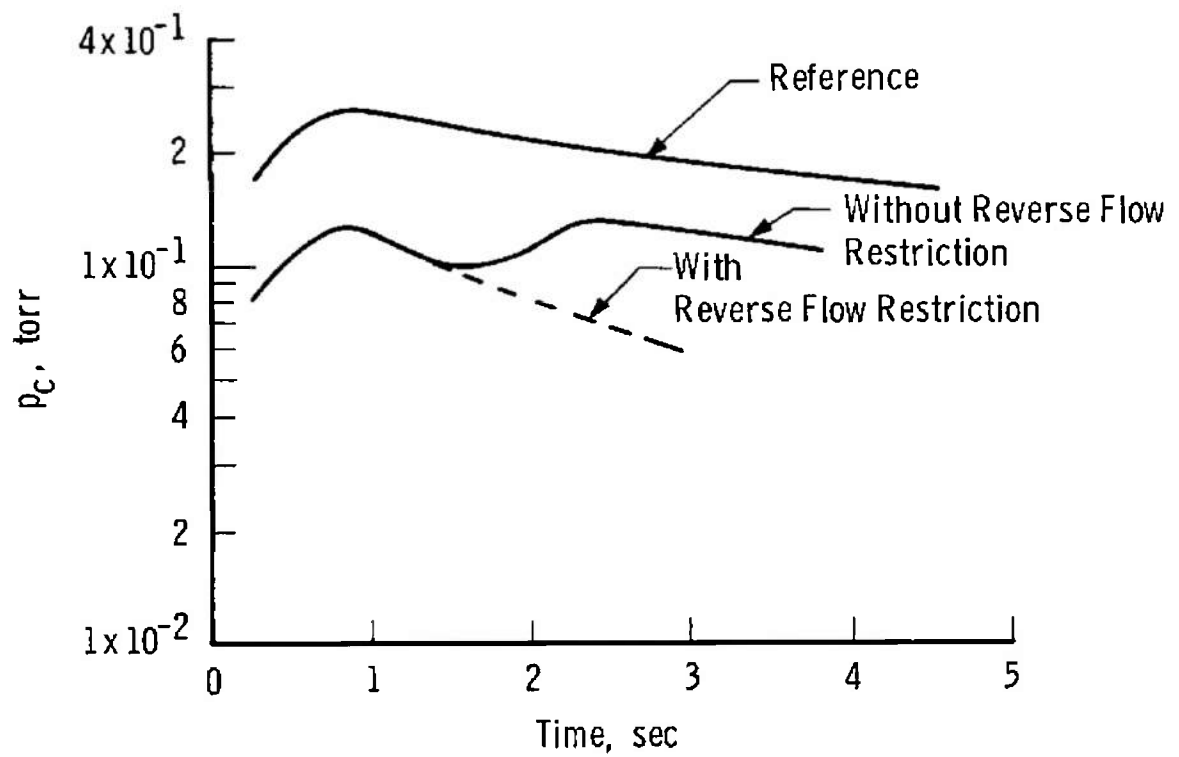


Fig. 22 Effect of Reverse Flow

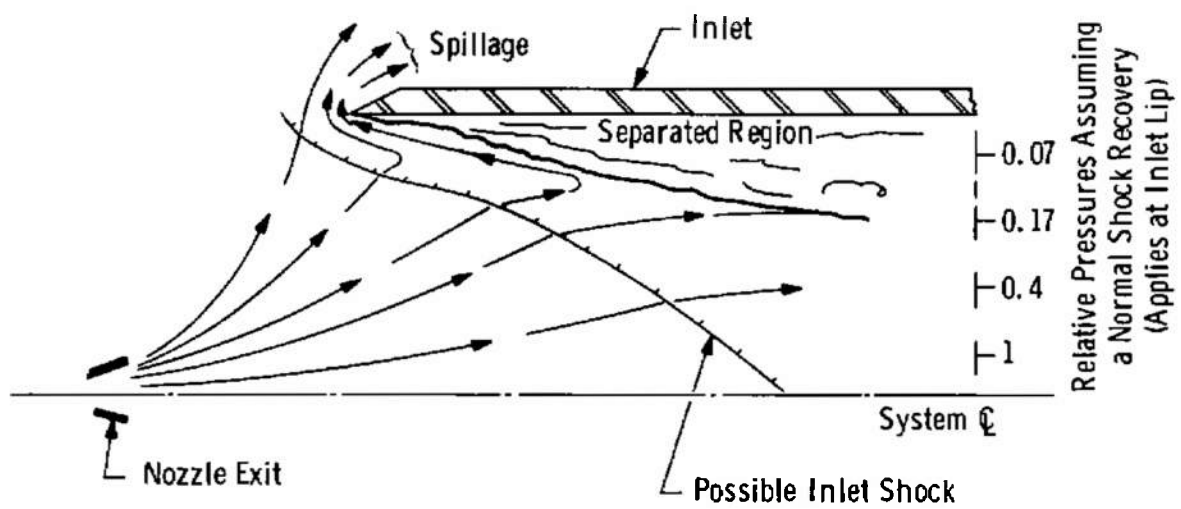


Fig. 23 Probable Inlet Flow Geometry

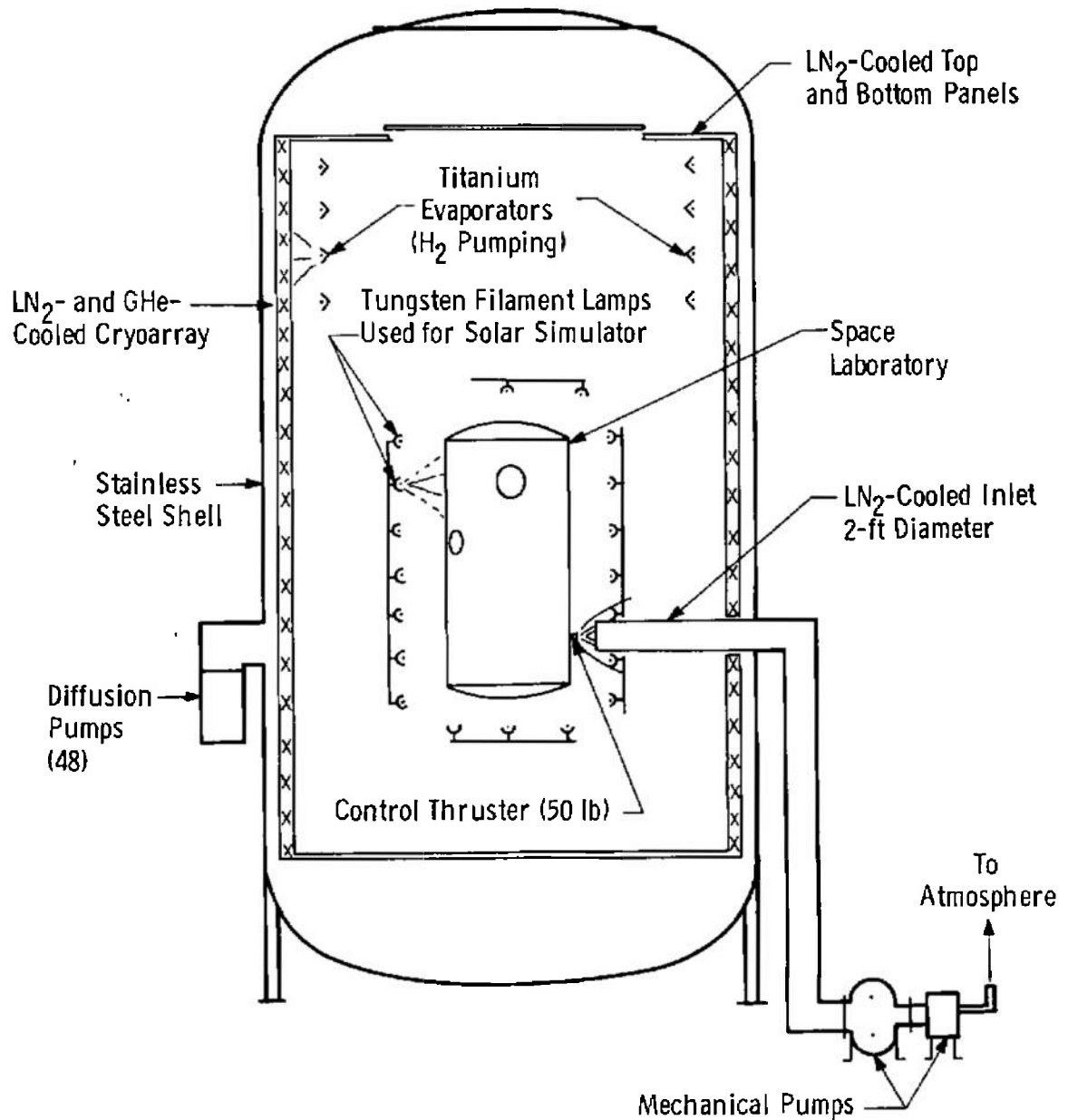


Fig. 24 Test Installation in a Large Space Chamber

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		2b GROUP N/A	
3 REPORT TITLE INVESTIGATION OF SCAVENGING SYSTEMS FOR REMOVAL OF ROCKET EXHAUST GASES FROM SPACE SIMULATION CHAMBERS			
4 DESCRIPTIVE NOTES (Type of report and inclusive dates) N/A			
5 AUTHOR(S) (Last name, first name, initial) Morris, J. A., and Anderson, D. E., ARO, Inc.			
6 REPORT DATE August 1966		7a TOTAL NO OF PAGES 49	7b NO OF REFS 12
8a CONTRACT OR GRANT NO. AF40(600)-1200 b Program Element 62405184 c Task 695001 d		9a ORIGINATOR'S REPORT NUMBER(S) AEDC-TR-66-108 9b OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A	
10 AVAILABILITY/LIMITATION NOTICES Qualified users may obtain copies of this report from DDC, and distribution of this report is unlimited.			
11 SUPPLEMENTARY NOTES N/A		12 SPONSORING MILITARY ACTIVITY Arnold Engineering Development Center Air Force Systems Command Arnold Air Force Station, Tennessee	
13 ABSTRACT Tests have been conducted on a rocket exhaust scavenging system as part of a program to develop new techniques for testing at simulated space conditions. Detailed plume characteristics were computed, and a model scavenging system was tested for both cold and hot gas rockets. Results indicate that for the firing of a typical small control rocket it will be possible to maintain 1×10^{-4} torr pressure in a space chamber.			

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
scavengers						
degassers						
rocket exhaust						
cold gas						
hot gas						
pressurization						
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